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FACILITIES AND OPERATIONS Final Report
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FINAL REPORT

LUNAR LANDING AND LAUNCH
FACILITIES AND OPERATIONS

JUNE 15, 1988

FLORIDA INSTITUTE OF TECHNOLOGY

150 WEST UNIVERSITY BLVD.
MELBOURNE, FLORIDA 32901-6988

LUNAR LANDING AND LAUNCH
FACILITIES AND OPERATIONS

Florida Institute of Technology

June 15, 1988

ABSTRACT

A preliminary design of a lunar landing and launch facility for a Phase III lunar base is formulated. A single multipurpose vehicle for the lunar module is assumed. Three traffic levels are envisioned: 6, 12, and 24 landings/launches per year. The facility is broken down into nine major design items. A conceptual description of each of these items is included. Preliminary sizes, capacities, and/or other relevant design data for some of these items are obtained. A quonset hut tent-like structure constructed of aluminum rods and aluminized mylar panels is proposed. This structure is used to provide a constant thermal environment for the lunar modules. A structural design and thermal analysis is presented. Two independent designs for a bridge crane to unload/load heavy cargo from the lunar module are included. Preliminary investigations into cryogenic propellant storage and handling, landing/launch guidance and control, and lunar module maintenance requirements are performed. Also, an initial study into advanced concepts for application to Phase IV or V lunar bases has been completed in a report on capturing, condensing, and recycling the exhaust plume from lunar launch.

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BACKGROUND

Florida Institute of Technology's NASA/USRA/UADP design team consists of senior level students from the following five departments: Chemical Engineering, Electrical Engineering, Mechanical/Aerospace Engineering, Ocean Engineering, and Physics/Space Sciences. Project administration consists of a Project Leader, Faculty Advisors from each department, and a Graduate Teaching Assistant (see Appendix 1). Students participating in the design team register for the course EGN 4001,2,3 "Special Projects in Space Systems Design I,II,III". Students may participate 1, 2, or 3 quarters depending on their departments curriculum requirements. A list of student participants for the 1987-88 academic year is given in Appendix 2.

The design course consists of an introduction to space systems design in the Fall Quarter. The introduction is composed of lectures by guest speakers from NASA and NASA contractors, Faculty Advisors, and the Graduate T.A. During the Fall Quarter the students are required to prepare a formal design proposal. During the Winter Quarter the bulk of the engineering design work is completed under the supervision of the Faculty Advisor from the students' major department. Regular "design meetings" are held where students present their work and discuss their projects with other students, Graduate T.A., and Project Leader. In the Spring Quarter, final reports and the summer conference presentation are prepared.

During the 1987-88 academic year Florida Institute of Technology presented the following papers:

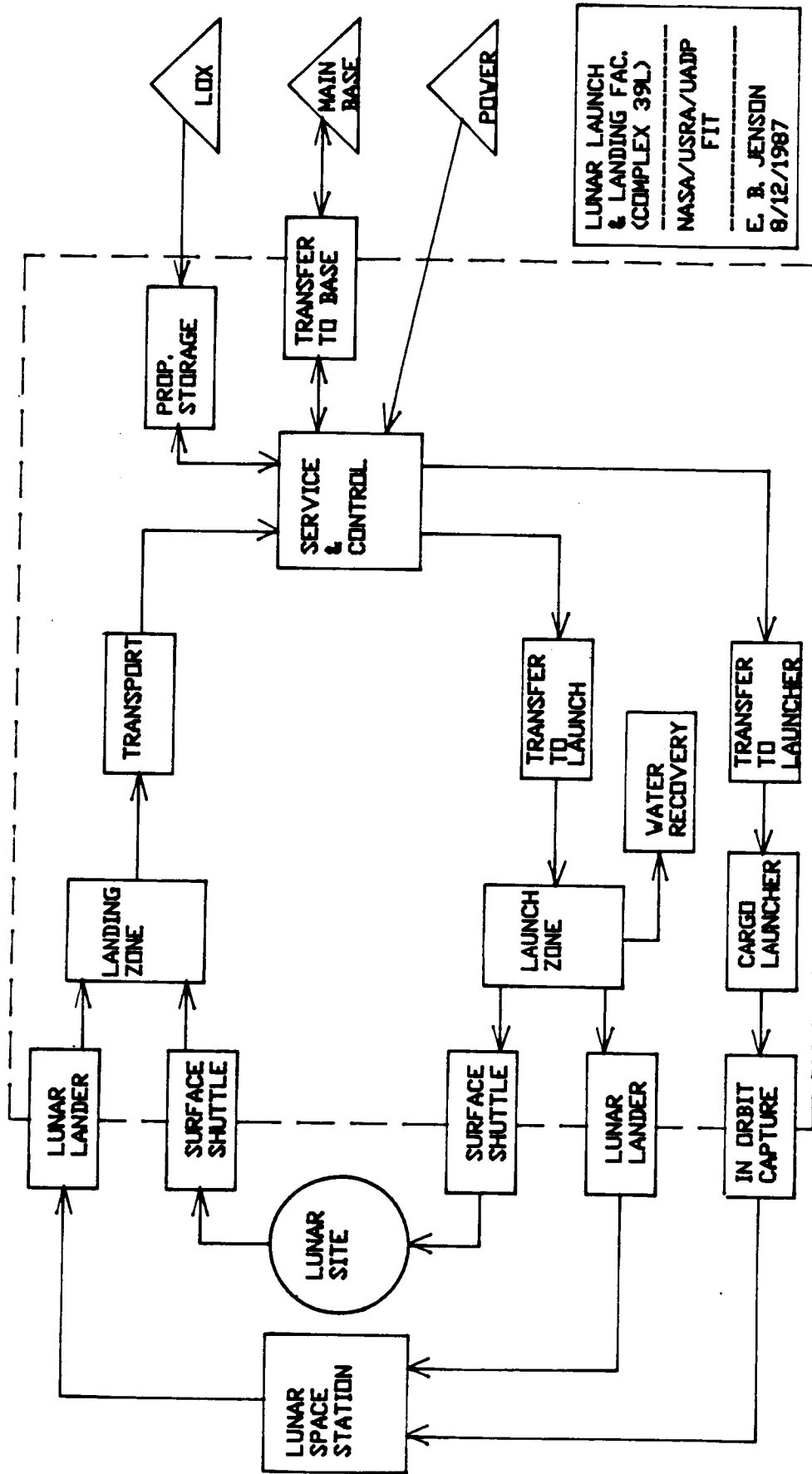
H.D. Matthews, E.B. Jenson, and J.N. Linsley
 "Preliminary Definition of a Lunar Landing and Launch Facility
 (Complex 39L)"
 NASA/AIAA/LPI Symposium: Lunar Bases and Space Activities of the
 21st Century; Houston, Texas
 Paper LBS-88-043; April 5 - 7, 1988

H.D. Matthews, E.B. Jenson, and J.N. Linsley
 "Lunar Landing and Launch Facility Concepts"
 Twenty-Fifth Space Congress; Cocoa Beach, Florida
 April 26 - 29, 1988

E.B. Jenson, D. Johnson, N. Geier, J. Linsley, and H.D. Matthews
 "Lunar Landing and Launch Facilities and Operations"
 Houston Section of the AIAA
 13th Annual Symposium; Houston, Texas
 May 11, 1988

A modified version of "Preliminary Definition of a Lunar Landing and Launch Facility (Complex 39L)" has been submitted for publication in the Lunar Bases and Space Activities of the 21st Century conference proceedings (see Appendix 3).

At the start of this year's design project a Lunar Landing and Launch Facility (Complex 39L) Systems Diagram was formulated. This initial diagram is given on the following page. Note that this diagram includes unmanned cargo launchers (mechanical or electromagnetic) and water recovery from lunar launch. Upon our decision to concentrate on a Phase III lunar base, we concluded that cargo launchers and water recovery systems should be removed. We believe that these advanced concepts are inherent to higher phases of lunar base development. This led to our present Lunar Landing and Launch Facility (Complex 39L) Systems Diagram (Appendix 3; Figure 1). With the exception of one report, all the design work completed this year is encompassed by our new systems diagram.



DESIGN WORK

This section consists of a short synopsis of students' reports, which are included in full in the Appendix section.

Appendix 3 is a paper entitled "Preliminary Definition of a Lunar Landing and Launch Facility (Complex 39L)" which has been submitted for publication in the Lunar Bases and Space Activities of the 21st Century conference proceedings. This paper provides the scope of this year's design project. Included are the basic assumptions made for a Phase III lunar base, transportation infrastructure, and baseline lunar module design. The Lunar Landing and Launch Facility Systems Diagram (Figure 1) and the Lunar Landing and Launch Facility Design Matrix (Figure 7) are presented. The lunar landing and launch facility is divided into nine major design items or areas, they are:

- [1.] Landing/Launch Site Considerations
- [2.] Shelter, Structure, Safety, Environmental Needs
- [3.] Landing/Launch Guidance, Communications, Computing Needs
- [4.] Lunar Module Surface Transport System
- [5.] Heavy Cargo Unloading/Loading Systems
- [6.] Personnel Unloading/Loading Systems
- [7.] Propellant Unloading/Loading Systems
- [8.] Vehicle Storage
- [9.] Maintenance, Repair, Test and Check-out Requirements

A description of each of these items is included, along with preliminary sizes, capacities, and/or relevant design data for some of these items.

Appendix 4 is a report entitled "Vehicle Assembly Tent: Preliminary Design". This report contains a preliminary structural design and thermal analysis for the main lunar module service depot. The structure is sized to accommodate four lunar modules and is in a quonset hut configuration. It is constructed of an aluminum frame with aluminized mylar panels. The main purpose of the tent is to provide a stable thermal environment for the lunar module and other equipment.

Appendix 5 is a report entitled "Maintenance and Safety Concerns for a Lunar Landing and Launch Facility". This report includes a preliminary investigation into the maintenance requirements for the reusable lunar module. A sample Lunar Module Turnaround Assessment is presented. Blast effect and radiation safety considerations are reviewed.

Appendix 6 is a report entitled "Bridge Crane Design". This report consists of a preliminary design of a heavy cargo unloading/loading bridge crane for unloading/loading cargo from the lunar module. The bridge crane will be located within the Vehicle Assembly Tent.

Appendix 7 is a report entitled "Bridge Crane Term Project".

This report presents a fundamentally different design approach to the heavy cargo unloading/loading bridge crane.

Appendix 8 is a report entitled "A Lunar-Based Propulsion System for a Launch and Landing Facility". This report includes a preliminary investigation into cryogenic fuel storage and handling systems at a lunar launch and landing facility. Safety considerations, storage vessels, and active cooling systems are addressed.

Appendix 9 is a report entitled "Water Recovery System". This report presents a launch pad design in which water vapor from combustion is captured and condensed for future use. Heat transfer calculations lead to an estimate for the condensation rate.

Appendix 10 is a report entitled "Lunar Landing Control System". This report consists of a preliminary design of a guidance and control system for the lunar module. Two landing modes are investigated. Landing pad requirements and environmental considerations for each landing mode are addressed. Surface based equipment sizing and placement is studied.

FUTURE WORK

Dr. Jerald Linsley will remain Project Leader for the 1988-89 academic year and the new Graduate T.A. will be Mr. Paul Borsetti from the electrical engineering department. A large recruiting effort has been undertaken this year and we expect a stronger design team consisting of top students from all departments.

Next years design project will consist of a more in-depth investigation of two of the main design items of this years project. The two items chosen for study are:

- (1) Storage and Handling of Cryogenic Fuels
- (2) Landing/Launch Guidance and Communications Systems

Note that these items still remain within the boundaries of our system as described in the Lunar Landing and Launch Facility Systems Diagram (Appendix 3; Figure 1). Next year's design team will produce more detailed designs than those of previous years.

ACKNOWLEDGMENTS

There are many individuals who have made significant contributions to this years design project. The first person we would like to acknowledge is our NASA contact, Mr. H. Dennis Matthews. Dennis has been a tremendous help in supplying us with current information from NASA and contractors working in areas similar to our own. He also co-authored three papers that we presented at various conferences throughout the year. His contributions to these documents were invaluable.

We would also like to thank Dennis' coworkers in the Future Project Office for their assistance, and especially for graciously letting us "take over" their computer room during our frequent visits to KSC.

We would like to thank our guest speakers from NASA and contractors who gave us a fantastic introduction to what the space program is all about. They took personal time to come to FIT to give excellently prepared presentations to our design class. They are:

Dr. William F. Huseonica	Pan-Am World Services, Inc.
Mr. Dennis Matthews	NASA/KSC
Mr. Bob Trujillo	EG&G
Mr. Kurt Buehler	NASA/KSC
Dr. Bill Knott	NASA/KSC
Mr. Bob Atkins	Martin Marietta
Mr. V. Leon Davis	NASA/KSC

We would like to thank the people at USRA and NASA Headquarters who have done a wonderful job of organizing the program and two superb conferences. Special thanks to Sue McCown for her assistance in getting us on NASAMAIL, and to Georgia Tech TA's Brice MacLaren and Gary McMurray for giving us someone to correspond with.

Special thanks to the Chemical Engineering Department at FIT for the many copies we made on their copy machine, and to the other support departments; copy center, audio-visual, interlibrary loan, and purchasing. We would also like to thank our Dean of Science and Engineering, Dr. Thomas Bowman, and the Physics/Space Science Department for giving us office space where such a commodity is generally in short supply.

APPENDIX 1

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Faculty ListNASA Contact:

Mr. H. Dennis Matthews Future Projects Office NASA/KSC

Project Leader:

Dr. Jerald N. Linsley Chemical Engineering

Faculty Advisors:

Dr. Ronald G. Barile	Chemical Engineering
Dr. Armand L. DiPare	Mechanical/Aerospace Engineering
Dr. J. Bruce Rafert	Physics/Space Sciences
Dr. Ronnal P. Reichard	Ocean Engineering
Dr. Michael H. Thursby	Electrical Engineering

Graduate Teaching Assistant:

Eric B. Jenson Chemical Engineering

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APPENDIX 2

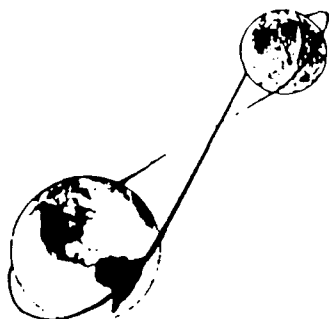
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Student Participants 1987 - 88

	<u>Dept.</u>
1. Aasland, Michael	Mechanical/Aerospace
2. Al-Hajri, Ali	Mechanical/Aerospace
3. Babbitt, Shannon	Mechanical/Aerospace
4. Biebesheimer, Richard	Mechanical/Aerospace
5. Blatt, Christopher	Chemical
6. Borsetti, Paul	Electrical
7. Buzzell, Judy	Mechanical/Aerospace
8. Carrier, Donald	Mechanical/Aerospace
9. Chavez, Elard	Mechanical/Aerospace
10. Chia, Yew	Mechanical/Aerospace
11. Dupper, Jeff	Mechanical/Aerospace
12. Garba, Samson	Mechanical/Aerospace
13. Geier, Nadine	Electrical
14. Hayes, Emily	Electrical
15. Hotaling III, Raymond	Mechanical/Aerospace
16. Hunter, Douglas	Mechanical/Aerospace
17. Johansen, Stig	Mechanical/Aerospace
18. Johnson, Doug	Mechanical/Aerospace
19. Karavitis, William	Mechanical/Aerospace
20. Mackenzie, Steve	Mechanical/Aerospace
21. McLoughlin, Peter	Mechanical/Aerospace
22. Poul, Roman	Mechanical/Aerospace
23. Scheer, Sharon	Mechanical/Aerospace
24. Scheunemann, Mark	Mechanical/Aerospace
25. Sciscente, Richard	Mechanical/Aerospace

26.	Sierra, Luis	Mechanical/Aerospace
27.	Ventura, Mark	Chemistry
28.	Wehbe, Fadi	Mechanical/Aerospace
29.	Zamudio, Ernesto	Mechanical/Aerospace

APPENDIX 3



LUNAR BASES & SPACE ACTIVITIES IN THE 21st CENTURY

April 5 - 7, 1988
Westin Galleria Hotel
Houston, Texas

PRELIMINARY DEFINITION OF A
LUNAR LANDING AND LAUNCH FACILITY
(COMPLEX 39L)

H. DENNIS MATTHEWS

FUTURE PROJECTS OFFICE
NASA/KENNEDY SPACE CENTER
KENNEDY SPACE CENTER, FL 32899

AND

ERIC JENSON
JERALD LINSLEY

DEPARTMENT OF CHEMICAL ENGINEERING
FLORIDA INSTITUTE OF TECHNOLOGY
150 WEST UNIVERSITY BLVD.
MELBOURNE, FL 32901-6988

Paper No. LBS-88- 043

Symposium sponsored by NASA, AIAA, the Lunar & Planetary Institute, the American Geophysical Union, the American Nuclear Society, the American Society of Civil Engineers, the Space Studies Institute and the National Space Society.

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PRELIMINARY DEFINITION OF A
LUNAR LANDING AND LAUNCH FACILITY
(COMPLEX 39L)

OUTLINE

- Abstract
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- II. Design Scope
 - A. The Lunar Module
 - B. The Lunar Base
 - C. Other Considerations
- III. Major Design Items
 - A. Landing/Launch Site Considerations
 - B. Shelter, Structure, Safety, Environmental Needs
 - C. Landing/Launch Guidance, Communications, Computing Needs
 - D. Lunar Module Surface Transport System
 - E. Heavy Cargo Unloading/Loading Systems
 - F. Personnel Unloading/Loading Systems
 - G. Propellant Unloading/Loading Systems
 - H. Vehicle Storage
 - I. Maintenance, Repair, Test and Check-out Requirements
- IV. Closure
- V. Acknowledgments
- References
- Figures

PRELIMINARY DEFINITION OF A
LUNAR LANDING AND LAUNCH FACILITY
(COMPLEX 39L)

H.D. Matthews

NASA/KSC

E.B. Jenson
J.N. Linsley

Florida Institute of Technology

ABSTRACT

A preliminary definition of a lunar landing and launch facility has been formulated. A permanently-manned lunar base is assumed. Also, a baseline lunar module is assumed. The major features of the facility are specified and major design areas are described. A design matrix is formulated. This design methodology can be extended to more detailed design cases.

I. Introduction

We have formulated a preliminary definition of a lunar landing and launch facility (LLLF or Complex 39L). We consider a phase III lunar base (Ref. 1,2). Without specifying specific lunar base scenarios, we envision three traffic levels: 6, 12, and 24 landings/launches per year. We have assumed a single, multipurpose vehicle for the lunar module, whose characteristics will be described below. The design and specification of the vehicle and of the lunar base are outside the scope of this study. However, these two items will have an impact upon those items that are considered within the scope of this study because of the interaction at the boundaries of our system. The scope of this study is graphically illustrated by the Systems Diagram of Figure 1. Here, major functions or facilities are represented by blocks in a block diagram. The dashed line represents the boundary of Complex 39L. This is a simplified version of this diagram. Obviously, other items could be included; e.g., lunar surface transportation and electromagnetic launchers. As previously mentioned, those items either on or outside the dashed lines which will have a significant impact upon the design of those items within the boundary will be discussed. Based upon this diagram, we have considered nine major design items or areas. These items are:

- [1.] LANDING/LAUNCH SITE CONSIDERATIONS
- [2.] STRUCTURE, SHELTER, SAFETY, ENVIRONMENTAL NEEDS
- [3.] LANDING/LAUNCH GUIDANCE, COMMUNICATIONS, COMPUTING NEEDS
- [4.] LUNAR MODULE SURFACE TRANSPORT SYSTEM
- [5.] HEAVY CARGO UNLOADING/LOADING SYSTEMS
- [6.] PERSONNEL UNLOADING/LOADING SYSTEMS
- [7.] PROPELLANT UNLOADING/LOADING SYSTEMS
- [8.] VEHICLE STORAGE
- [9.] MAINTENANCE, REPAIR, TEST AND CHECK-OUT REQUIREMENTS

First, we provide a general, conceptual description of each of these items. Then, we have obtained preliminary sizes, capacities, and/or other relevant design data for some of these items. This constitutes an obviously preliminary design study of a phase III lunar landing and launch facility.

II. Design Scope

A. The Lunar Module

We assume a baseline transportation system (Ref. 3). The transportation infrastructure (Figure 2) will consist of a low earth orbit (LEO) space station, a low lunar orbit (LLO) space station, orbit transfer vehicles (OTV), lunar modules, and a lunar landing and launch facility (LLLF or Complex 39L). The LLLF is part of the total lunar base. Both the OTVs and lunar modules will be reusable with no expendable vehicles considered. For the baseline transportation system all vehicle propulsion systems use hydrogen/oxygen (H/O).

The basing scenario includes the space station in LEO to provide servicing, payload accommodation, and propellant supply. We use propellant to refer to liquid oxygen as well as liquid hydrogen. A similar basing node located at LLO will be needed as a propellant storage depot, a payload transfer depot, and for the servicing of either OTV or lunar module systems. The final basing node will be at the lunar surface base (Complex 39L) and will have propellant storage, payload transfer, and lunar module servicing capabilities.

For the flight from LEO to LLO the OTV will carry a manned capsule, payload, and propellant for the lunar module and for its return to LEO. For the flight from LLO to LLLF the lunar module will carry a manned capsule, payload, and propellant for its return to LLO. Unmanned OTVs and lunar modules in which the manned capsule is replaced with an increased payload can also be used. For this study we assume a preliminary baseline lunar module design.

The baseline lunar module is a reusable, two-engine vehicle

capable of delivering 15.9 MT (manned capsule plus payload) to the lunar surface from LLO and returning to LLO with an equivalent payload (Ref. 3). The manned capsule will have the capability of transporting 6 people (2 pilots and 4 passengers). The vehicle components are modeled after the Centaur D-1T. The combustion chamber operates at a pressure of 13800 kPa (2000 psia) and a temperature of approximately 3600 K (6000 °F). The thrust per engine in a vacuum is 33,400 N and the gas exit velocity is approximately 2,300 m/s. The oxygen/hydrogen mixture ratio is 5.5 on a mass basis. The propellant tanks are cylindrical with elliptical heads. The landing gear mass for a lunar module is assumed to be 5% of the total landing mass on the moon. Lunar module characteristics are summarized in Figure 3. The vehicle lifetime with minimum maintenance is estimated to be forty flights.

Dimensions for the lunar module were estimated from the weight, engine size, and by making comparisons to the lunar excursion module (LEM) used in the Apollo missions. The height, which is estimated at 10 meters, is the vertical distance from the footpads to the top of the vehicle. The diameter, which is estimated at 13 meters, is the distance from footpad to diagonal footpad. The lunar module has four footpads, which gives a distance of 9.2 meters from footpad to adjacent footpad. This vehicle is consistent with recent studies by Johnson Space Center (Ref. 4).

The lunar module is assumed to have a "modular" design. That is to say that main subsystems (propellant tanks, engines, cargo modules, manned capsule, etc.) can be easily removed and replaced. For example, if a propellant tank is damaged it will be removed and replaced with a new or repaired tank. The replacement tank will come from either an equipment inventory or from another vehicle. Due to the expense and hazards of extra vehicular activity (EVA) it will be advantageous to make maximum use of robotics to perform the required lunar surface tasks (Ref. 5). However, we believe that at this stage in lunar base development the tasks will be diverse and complex enough so that most repairs will need to be made by personnel wearing space suits. This requires special design consideration in an attempt to facilitate the person making the repairs.

B. The Lunar Base

For this study we assume a phase III lunar base (Ref. 1,2). This is a permanently occupied facility in the time frame of 2005 - 2009. The human population will range from approximately 10 to 30 during this time period. The base will emphasize both scientific research and in-situ resource utilization. The following is a partial list of the research projects that will be undertaken: CELSS experiments, cosmic and gamma ray observatories, radio telescope, life sciences research, and search for extraterrestrial intelligence (Ref. 6,7). These are listed to give the reader a general idea of some of the payloads

that must be brought to the lunar surface via Complex 39L. It will be desirable to make use of resources available on the moon in an attempt to minimize the required earth launch mass (ELM). This will have a significantly large impact on the cost of the project. The main resources that can be derived from lunar mining are: Oxygen, Metals, Glass, and Ceramics. These materials will be used in the operation and construction of the lunar base. It should be pointed out that resource utilization is inherent to later stages of lunar base development (Phase III and up). The baseline case for this study assumes minimal use of lunar resources.

The lunar base can be broken down into several sub-facilities. These will include the habitat modules, various lunar production facilities, nuclear power facility, and lunar landing and launch facility (Complex 39L). The habitat modules will be used for laboratories and offices as well as for habitation. Lunar base headquarters and the majority of the lunar population will reside in them. The other sub-facilities will be designed to perform their individual functions and to support a small operating crew for short durations. A preliminary plot plan for Complex 39L is given in Figure 4. We now discuss the interaction between Complex 39L and the other sub-facilities; ie. the boundaries of our systems diagram.

It is assumed that habitat and laboratory modules similar to those used in the LEO and LLO space stations will be used on the moon. The modules will be covered with lunar regolith for radiation protection (Ref. 8). The increase in the number of inhabitants must be accompanied by an increase in the number of habitat modules and therefore an increase in the size of the base. The landing, transportation, and assembly of habitat modules will be an ongoing activity at the lunar base. An increase in the number of inhabitants will also call for an increase in consumables (water, food, oxygen, etc.) required. We assume a baseline crew rotation of six months. With a population of 30 this will require 15 lunar module flights per year to meet the crew rotation schedule.

The construction and operation of a lunar liquid oxygen (LLOX) production facility will occur during phase III of lunar base development (Ref. 1,2). LLOX will initially be used for habitation on the moon and as the oxidant for lunar modules. The long range goal will be to export LLOX to LLO and LEO space stations and to use it as the oxidant for OTVs and other space vehicles. As the base and the amount of LLOX exported increases so must the capacity of the LLOX plant. When the capacity must be increased another LLOX production module must be landed, assembled, and put on-line. Each LLOX production module is assumed to have a standard production capacity, and the LLOX plant will be made up of these modules operating in parallel. LLOX must be transported from the LLOX plant to the storage tanks at Complex 39L. We envision that LLOX will be transported across the lunar surface with a vehicle analogous to a terrestrial tank truck.

The power requirements of the lunar base will increase as the production capacity and number of inhabitants increases. Nuclear energy will most likely be the main source of electric power. The power available is estimated to range from 1 to 10 MW over the given time period (Ref. 1). To meet the increasing demand nuclear reactors and supporting elements must be landed, transported, and assembled during the expansion of the base. Other supplemental energy forms such as solar energy will also be used with similar installation requirements.

The lunar base sub-facilities will be separated from Complex 39L by a specified distance. This distance will depend on safety considerations and the site of the base. The potential of an explosion, large navigation errors in landing, and rocket engine blast will warrant separating Complex 39L some distance from other facilities. The ideal site for Complex 39L will be a large expanse of flat rock or an area where the surface can be easily prepared for landing and launch activities. The habitat modules will require a site that contains a large amount of loose regolith for radiation protection. This site selection criteria might make it necessary to separate the facilities further than would be required by safety considerations alone. For an estimate we are assuming the distance between Complex 39L and the lunar base to be from 3 to 5 kilometers (Ref. 9).

A surface transport vehicle will be required to transfer payload and personnel between sub-facilities. A payload transfer system at Complex 39L will be used to transfer payloads from the lunar module to this surface transport vehicle. The vehicle will contain a pressurized compartment for personnel transport (Ref. 1). A prepared surface will be required if a wheeled vehicle is used. Prepared roadways will connect the various sub-facilities.

C. Other Considerations

In this section we discuss a number of items which are considered "outside-the-scope" of the design; i.e., outside the dashed boundary of Complex 39L as indicated on Figure 1. In this consideration of items being either inside or outside the scope of design follows good design practice (Ref. 10).

For purposes of our design, we consider only manned lunar modules. We realize that design requirements for manned vehicles as opposed to unmanned vehicles are considerably more stringent and that there will be an effort to use unmanned vehicles to the greatest extent possible. Manned vehicles will also impose more stringent constraints on the design of Complex 39L. The only area where unmanned vehicles will impose more requirements is in the guidance and communications area. These increased requirements can be considered within the margin of error in this preliminary analysis. Thus, the consideration of only manned lunar modules will impose conservative design requirements on the

landing and launch facility.

It is generally accepted that a high degree of utilization of automation and robotics technology will be used in lunar base activities. While recognizing this, we take many of our design concepts from current technology that has not yet experienced automation or robotics technology advances. Again, this is done partially in the interest of obtaining a conservative design. We also consider that the highly automated and roboticized facilities will be heavily interspersed with rather low technology devices. Designers of lunar base equipment should look to the seven basic machines of elementary physics for initial design concepts.

A number of advanced concepts have been omitted from this study. We mention two of these. Electromagnetic launchers are not considered in this Phase III design. This is a popular concept in discussions of lunar base design. Another less popular concept that we have considered but omitted from this study is the design of a landing and launch pad from which we attempt to recover water vapor from the exhaust plume. We consider this interesting concept to be beyond Phase III.

We have borrowed heavily from a recent Eagle Engineering report on Lunar Landing and Launch operations. While this study is concerned mostly with Phase I and II operations, many of the concepts presented therein (Ref. 9) are valid for Phase III.

III. Major Design Items

A. Landing/Launch Site Considerations

The lunar module will touchdown vertically on a specified zone (landing/launch pad). For lunar module transportation requirements and dangers from engine blast effects it is desirable to have a paved landing/launch pad. Loose particles on the pad can become dangerous projectiles in the presence of engine blast from the lunar module. With a paved pad this problem is greatly diminished. In this study we assume that the same pad will be used for both landing and launch. The ideal site would be a large expanse of flat rock which could be cleared of lunar dust and used without any other preparation. The next best case would be to locate an area where the lunar regolith could be excavated to uncover a suitable hard rock site. From lunar surface studies it is unlikely that such an ideal site can be found (Ref. 11). Alternatively, it becomes necessary to prepare a landing pad by other means. Concrete from lunar regolith, lunar gravel, or bags of lunar regolith are three possibilities. For the remainder of this study we assume a paved landing pad. It should also be noted that large amounts of lunar regolith must be available near the landing site to provide radiation protection for habitat modules and to create blast barriers if necessary.

The landing pad (Figure 4) will be circular with a diameter of 50 meters (approximately four times the diameter of the lunar module). This figure was arrived at by making comparisons with terrestrial vertically landing vehicles. A similar figure (100 m) was arrived at independently by Eagle Engineering from consideration of cruise missile technology. Due to possible navigation errors and harmful blast effects a circular area of approximately 250 meters from the center of the landing pad will be cleared of large rocks and equipment. The landing pad will be marked with lights similar to a terrestrial airport to aid the pilots of the lunar module, and television cameras for the controllers in the communication and control facility. This will be the only equipment within the 250 m area during landing or launch. This equipment must be capable of handling any engine blast effects that might occur. This would include replacable lens covers on cameras. The number of landing/launch pads will depend on the flight schedule and the time required for pad maintenance. Figure 4 shows one pad, though more may be required.

B. Shelter, Structure, Safety, Environmental Needs

It is assumed that the lunar module will spend a significant period of time on the lunar surface. This could be from 2 weeks to 2 months. It will be desirable to control the temperature of the vehicle by removing it from direct sunlight. This will decrease the boil-off of cryogenics and also provide a constant thermal environment. The module will be serviced by personnel wearing space suits. A shaded environment will decrease both visibility problems and life support requirements (Ref. 5). Other equipment such as robots and tools will also benefit from a stable thermal environment.

We propose the use of a quonset hut tent-like structure (Figure 5). This structure will be referred to as the vehicle assembly tent (VAT). The facility will be large enough to contain four lunar modules. The dimensions are 50 m long, 36 m wide, and 18 m high at the center line. Entrances, 15 m high and 16 m wide, will be located at each end of the structure. A framework will be constructed of a material such as 2014-T6 aluminum (Ref. 12). Highly reflective panels made of a mylar/evaporated aluminum laminate will shield equipment inside the VAT from incoming thermal radiation. These panels are expected to reflect approximately 90% of the thermal solar spectrum (Ref. 13). Other panel materials and laminates are being investigated. Initial calculations with one layer of panels give a surface-level temperature inside the VAT of approximately zero degrees celsius during the lunar day. It was found that using two layers of panels separated by 0.1 m gave a decrease in surface temperature of only 8 °C. We make the assumption that this relatively small decrease in temperature will not warrant the additional ELM or construction and maintenance time required for a second layer of panels. The panels will be sized so that a single person wearing a space suit

could replace one easily. An initial size of 2 m X 2 m will be assumed. The panels will attach to the frame at points approximately every 0.5 m. Approximately 850 panels will be required. When a panel has degraded or been damaged it will be replaced with a new or refurbished one. Movable flaps will be used to cover the entrances at each end of the structure. They will serve to block glare and possible particles from engine blast. These will be made of the same material as the panels and will function similar to a typical stage curtain. They will cover an area of 240 square meters at each end. A total mass of approximately 10 MT has been derived for the proposed structure.

Most servicing and unloading/loading operations will be performed in the VAT. Artificial lighting, etc. must be provided where men are working. This is not a pressurized facility and personnel must wear space suits. This facility will not block radiation that is potentially dangerous to humans. The amount of time humans can work in this environment will be limited (Ref. 14). Personnel will be able to work in the VAT approximately 33 hours per week without exceeding earth-based exposure limits (5 REM/year). All surface activity must be discontinued during periods of solar maximum (large solar flares). It may be determined that all surface operations will be best performed during the 14 day lunar night. Electronic devices will also be affected by high doses of radiation in the form of both hardware and software upsets. Electronic equipment should be specially designed for lunar applications.

One of the major environmental problems is damage caused by engine blast during lunar landing and launch. The facility is located a distance of somewhat greater than 250 m from the nearest landing pad, which should be sufficient to eliminate any large particles (greater than 0.5 mm). Delicate equipment should be protected during landing and launch operations by use of a shield or protective blanket. For safety considerations all personnel on the surface within approximately 3 to 5 km of the pad will be required to remain behind a protective barrier until the landing or launch operation is complete. It is possible that a large barrier could be constructed between the pad and the VAT. This could be a pile of lunar regolith or a wall made from bagging lunar regolith.

C. Landing/Launch Guidance, Communications, Computing Needs

The lunar module will be manually controlled by two pilots. Assistance will be provided by a surface communications and control facility similar to that of a terrestrial airport. It has been determined that currently available terrestrial navigation systems can be applied to achieve high degrees of landing and positioning accuracies (Ref. 9). Onboard systems will utilize terrain matching systems during periods when the base is out of view. Terrain matching radar technology is used with high degrees of accuracy in modern cruise missiles. An accurate map of lunar surface features will be required. When

the lunar base is in view of the lunar module, surfaced based transponders will be used. Transponders will be placed on the surface to give high triangulation resolution. They should be detectable by the lunar module radar system at a distance of 200 km. It is assumed that five transponders will be used. Three transponders will be placed 120 degrees from each other at a distance of 100 m from the center of the landing pad. Two other transponders will be placed 1.5 km downrange and 1.5 km crossrange respectively. A surfaced based radar system consisting of a dish approximately one meter in diameter will be used to track the transponder on the lunar module. This will enable the operators in the communication and control facility to follow the lunar module and to abort unmanned flights if necessary.

The communications and control facility will be located at Complex 39L. Habitat modules will be used to house the equipment and operators. Approximately two people will be required to operate the facility. The operators will be responsible to coordinate landing/launch operations. It is envisioned that during heavy processing times it will be necessary for people to live at the facility. Temporary living arrangements will be provided. The facility will also be used to do minor repairs of lunar module components that require a shirt sleeve environment.

D. Lunar Module Surface Transport System

The lunar module will be transported from the landing/launch pad to the VAT. We envision the use of a battery powered lunar forklift. A dolly will be placed under each footpad and a harness will be attached to the lunar module or dollies. The lunar module will then be towed by the forklift into the VAT. The forklift and the dollies will function best if the towing surface is paved and level. It is assumed that the vehicle will be operated by a person in a space suit, though this is an area with potential robotic application that should be investigated. This same system will be used for transporting lunar modules to vehicle storage.

E. Heavy Cargo Unloading/Loading Systems

Heavy cargo items such as habitat modules, construction equipment, nuclear reactors, and LLOX production modules will be transported to the lunar base on a regular basis. These items will be attached to the lunar module and may or may not be stored in containers. The lunar module will be transported into the VAT fully loaded with payload. Once in the facility the module will be unloaded. We envision the use of a bridge crane. The crane will encompass an area of 15 m by 30 m at a height of 15 m. For a preliminary design we assume a maximum load of 45 MT. We design the center beam to have a deflection less than 0.05 m with the maximum load applied at its center point. It was found that a standard 24 X 62 wide flange beam constructed of 4340 low

carbon steel will meet these design requirements (Ref. 15). This is a baseline design and other construction materials are being investigated. We assume that the entire structure will be constructed of the same members. This gives a total crane mass of approximately 20 MT.

Operations will begin by detaching (unstrapping) a payload from the lunar module. The crane will then be positioned and attached to the payload. The payload will be lifted, transported away from the lunar module, and lowered onto either a lunar transport vehicle, a dolly, or the lunar forklift. If the transport vehicle is not available to take the payload directly to its destination, then the payload will be transported to cargo storage at Complex 39L to await further processing. The lunar forklift will be used to tow the dollies or to lift the payloads directly. Complex 39L cargo storage will be a separate tent structure similar to the VAT and located nearby (Figure 4). The lunar module will be loaded in an inverse manner.

F. Personnel Unloading/Loading Systems

We envision three modes of personnel unloading/loading. The first requires EVA and is similar to that used in Apollo missions. The personnel will don space suits and exit the lunar module by climbing down a ladder that is attached to one of the module legs. This can be done either on the landing/launch pad or in the VAT. If the personnel exit or enter the vehicle on the pad they must either walk or be transported to or from the pad. It would be advantageous for several reasons to have the personnel exit the vehicle after it has been towed to the VAT. The main reason is a decrease in the amount of total EVA time required.

The second mode of unloading/loading is for the personnel to remain in the module until it has been transported into the VAT. They will then disembark into the pressurized compartment of a lunar surface transport vehicle. The personnel will be transported to the habitat modules or wherever their final destination may be. This is a "shirt sleeve" transport operation where space suits are not required. This mode will require a pressurized transport and an airlock mechanism to connect the two vehicles.

The third mode of unloading/loading is again for the personnel to remain in the lunar module until it has been transported into the VAT. Here, the manned capsule of the lunar module will be detached, lifted by the bridge crane, and either placed on a dolly or on a lunar surface transport vehicle. The entire manned capsule will then be transported to the habitat modules where the personnel can disembark through an airlock. This is again a "shirt sleeve" operation where EVA is not required. A separate pressurized transport will not be required as in the second mode. This example illustrates the integration (modularity) that we believe is necessary for a successful lunar

base.

G. Propellant Unloading/Loading Systems

The lunar module will land at Complex 39L with some propellant remaining in its fuel tanks. Assuming no LLOX is available this will be all the hydrogen and oxygen required for the return flight to LLO. The propellant can either be left in the fuel tanks or transferred into propellant storage tanks. If boil-off from the fuel tanks is large then it would be preferable to store the cryogens in larger tanks with active cooling systems. Hydrogen and Oxygen storage tanks will be located at Complex 39L. We assume that active cooling systems will be used. The cooling systems will be designed to achieve a specified maximum boil-off.

The storage requirements are set by the number of people residing at the base. As a design criteria we require that enough propellant be stored to evacuate the entire lunar base population. For a population of 30 this would require storage of approximately 150,000 kg of oxygen and 30,000 kg of hydrogen. If one spherical tank is used to store each cryogen this would require tank diameters of roughly 6.0 m and 9.0 m for oxygen and hydrogen, respectively. Multiple tanks of differing geometries may be used.

The storage tanks and pumps will be located in a separate tent near the VAT. This tent is referred to as the fuel inventory tent (FIT), Figure 4. The lunar module will be defueled/fueled by removing the propellant tanks from the module with the bridge crane, placing them on dollies, and transporting them to the propellant storage tent in the same manner that cargo is transported. All propellant transfer operations will be performed in the FIT.

H. Vehicle Storage

A long term vehicle storage area will be provided at Complex 39L (Figure 4). This will be an area near the VAT that has been cleared of large objects. It is assumed that some surface preparation will be required for transportation purposes, but it is unlikely that the paving requirements will be as stringent as for the landing pad. At this phase in lunar base development we envision an area large enough to contain 6 lunar modules (approximately 1000 m²). With an increase in the lunar module fleet and landing/launch rate this area will need to be enlarged.

The lunar module will be transported to vehicle storage if it has been damaged beyond repair, exceeded its operational life, or will not be used for a long period of time. The lunar modules will have been defueled prior to storage. The module will be towed by the lunar forklift to a storage location as previously discussed. A dome tent will then be pitched over it. This will be

a tent made up of the same panels used in the VAT attached to a metal or composite support frame. The tent is used to achieve a constant thermal environment and to protect the module from potential particle bombardment. The tents must be designed to enable assembly by a few personnel in space suits.

Lunar modules in vehicle storage will essentially be used for cannibalization. We assume that lunar module components will have different operational lives. Some components will still be operational when the vehicle as a whole is not. Working components from vehicles in storage will be used to repair operational vehicles in the VAT.

I. Maintenance, Repair, Test and Check-out Requirements

The lunar module is a reusable vehicle and will require regular maintenance with each flight. Vehicle maintenance will include but is not limited to: vehicle test and check-out, lubrication, recharging of environmental systems, recharging or replacing batteries, regenerating fuel cells, modular parts replacement/repair, and system modifications. Unlike presently operated reusable terrestrial space vehicles, the lunar module should require minimal maintenance. For our highest frequency flight schedule (24 flights/year), the lunar module turnaround time will be two weeks (14 days). For a baseline case we assume that routine maintenance will be performed by two personnel. However, more manpower will be required if a significant problem develops or if major system alterations are required.

As previously described we envision a modular vehicle design. If a lunar module component is damaged it will be replaced by a new component or one from another vehicle. Due to limitations imposed by EVA, design work needs to be done to insure that connections can be made easily by personnel in space suits. Also, these connections will present potentially weak links and will require inspection prior to launch.

When a lunar module component malfunctions or is damaged, it can either be repaired or "scrapped". Repairs can be performed on-site, at one of the space stations, or on earth. In many cases it will be desirable to repair the component at Complex 39L. Lunar module mechanics will be constrained by space suits making detailed repairs difficult if not impossible. Smaller components (computer systems, circuit boards, etc.) can be brought into the Communications and Control Facility to be worked on in a shirt-sleeve environment. Detailed repairs on larger components (engines, fuel tanks, etc.) will require a large pressurized facility. We believe that such a facility is beyond a Phase III lunar base, and repairs of this nature will not be possible at Complex 39L. We assume that there will be tasks, such as replacing bolts and rivets, that can be performed adequately by personnel in space suits. Specially designed tools and equipment will be required.

We envision that in many cases the repair requirements will be overly involved and the vehicle will be unusable. If this is the case the vehicle will be towed to vehicle storage as previously described and a new vehicle sent from earth to replace it. Working components on the stored lunar module will be cataloged for use in the repair of operational vehicles. Spare parts equivalent to at least one lunar module should be stored at Complex 39L.

We now discuss the routine maintenance, repair, test and check-out procedures that will be followed for each lunar module flight. We identify four main procedures, they are: Initial Safing, Postflight Servicing, Lunar Module Modification, and Preflight Servicing. The following is a preliminary description of some of the operations that will be performed during each of our identified main procedures.

Initial Safing will include: transportation of the lunar module into the VAT, unloading or draining fuel tanks, attachment of ground power and purge lines to the lunar module, purging main engines and fuel lines to remove possible moisture resulting from hydrogen/oxygen combustion, and unloading payloads. Also, the lunar module crew will disembark sometime during the initial safing procedure. This is a preliminary list of required operations that can easily be expanded upon.

After Initial Safing is complete, postflight troubleshooting begins to determine anomalies which may have occurred during launch, spaceflight, or landing. An umbilical containing electrical, communication, instrumentation, and control lines is connected to the vehicle. Visual and electrical inspections are performed on the lunar module. Along with postflight inspection, routine servicing will include: lubrication, recharging of environmental systems, recharging or replacing batteries, regenerating fuel cells, and other required routine maintenance tasks.

Lunar module modifications will then be made if necessary. Modifications will include: replacing damaged components that were discovered during the postflight inspection, adding or removing equipment necessary to meet future mission requirements, and replacement of outdated hardware with new designs to enhance vehicle performance. Lander modifications, if extensive, can be performed over a long period of time while the craft is in vehicle storage. However, many modifications will be performed in parallel with routine servicing.

The lunar module will then be prepared for launch. Preflight Servicing will include: installation of flight supplies and payload, attachment of fuel tanks, loading of personnel, visual and electronic check-out of lunar module systems, detachment of ground umbilicals, and transportation from the VAT to the launch pad. Again, other operations can be included.

For our four main procedures we develop a sample Lunar Module

Turnaround Schedule (Figure 6). This figure is based upon preliminary estimates for the time requirements for the four main procedures. We recognize that this, though an overly simplistic representation, is the most detailed definition of reusable lunar module servicing to date.

IV. Closure

We have presented short descriptions or specifications of our nine designated design items. The next stage in our design process is to determine preliminary estimates for the major resource requirements of our system. We identify three major resources to be mass, power, and manpower. The cost of the lunar base will be directly related to the resource requirements. While mass and power requirements can generally be determined by standard engineering methods, assessments of manpower requirements can be difficult. One assumption that is made is that all EVA operations will be undertaken by at least 2 personnel. This is a safety consideration which mimics the "buddy system" that is used in scuba diving.

A Design Matrix has been developed (Figure 7). This is a matrix composed of the nine major design items as rows. The specification, mass, power, and manpower are the columns. We recognize three main resource requirement areas, they are: Construction, Operation, and Maintenance. The Construction area represents the resources that will be required during the construction phase of the landing and launch facility. This will include but not be limited to: clearing a site of large debris, landing/launch pad preparation, pitching of various tents, and assembly of cranes and other structures. The operation area represents the resource requirements for the "steady-state" operation of the facility. And, the maintenance area represents the facility maintenance requirements; e.g., refurbishment of landing/launch pads, tent structures, and other hardware (forklifts, cranes, etc.). A matrix element is "checked-off" when the design work corresponding to that element has been completed. Obviously, more rows and columns can be incorporated into this design methodology as the need for greater and greater design detail is required.

V. Acknowledgments

This work was done under the auspices of the NASA-sponsored University Advanced Design Program through the University Space Research Association, i.e., this has been primarily a senior class engineering project. The Future Projects Office at NASA-KSC has been most helpful. Those who deserve the most thanks are

our students.

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6. Lunar Module Turnaround Schedule
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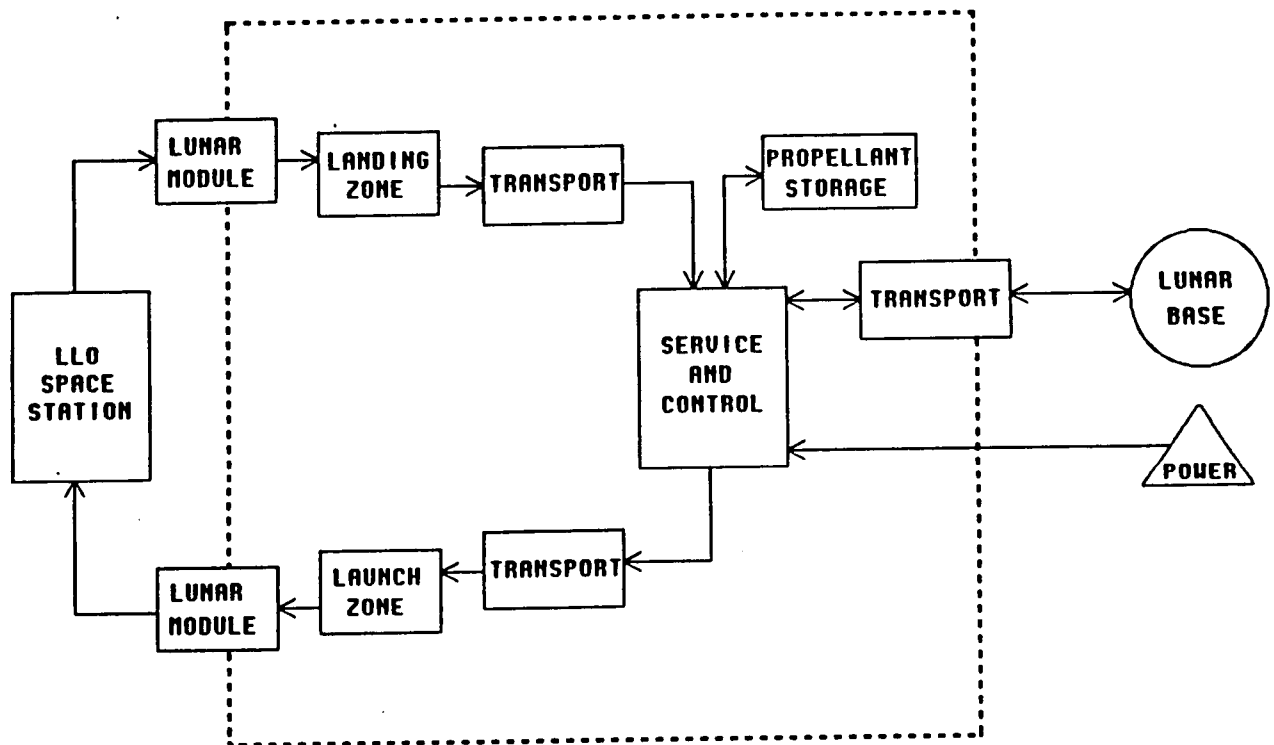


Figure 1: Lunar Landing and Launch Facility

(LLLF or Complex 39L) Systems Diagram

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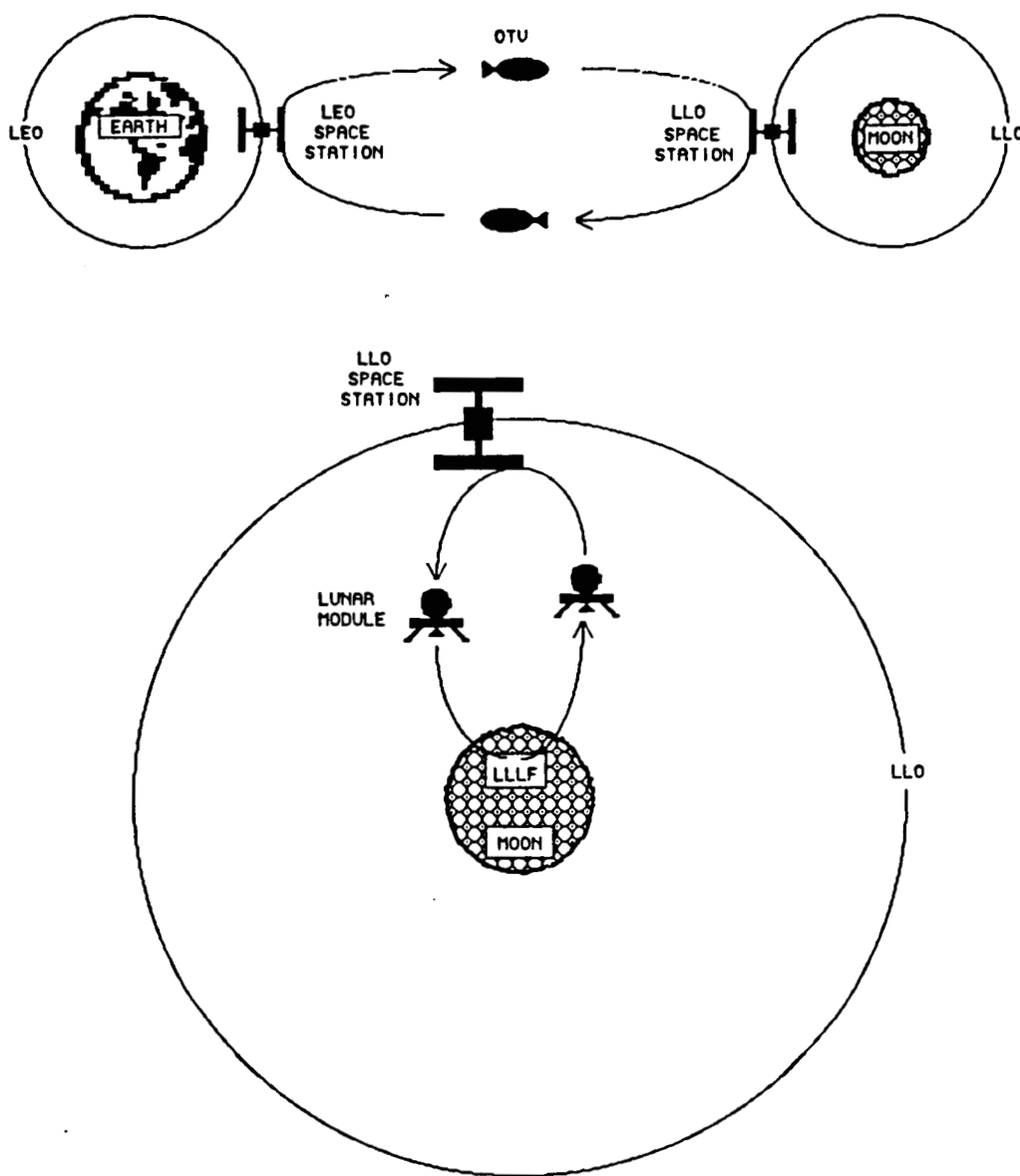


Figure 2: Earth -- Moon Transportation Infrastructure

Figure 3: Baseline Lunar Module Design

ENGINE DATA:

Isp (s) :	470
Number of engines :	2
Thrust per engine (N) :	33400
Mass of each engine & its thrust structure (kg) :	95
Mass ratio Oxygen/Hydrogen (kg) :	5.5

APPROXIMATE MASS (kg):

Dry mass :	1000
Landing gear mass :	1800
Oxygen tank mass :	100
Hydrogen tank mass :	<u>400</u>
TOTAL MASS :	3300

APPROXIMATE PROPELLANT CAPACITY (kg):

Oxygen :	21000
Hydrogen :	<u>4000</u>
TOTAL PROPELLANT CAPACITY :	25000

APPROXIMATE DIMENSIONS (m):

Height :	10
Diameter :	13

APPROXIMATE PAYLOAD CAPACITY (kg):

Maximum payload capacity :	15900
Liftoff payload capacity :	15900
Manned capsule (crew of 6) :	6900

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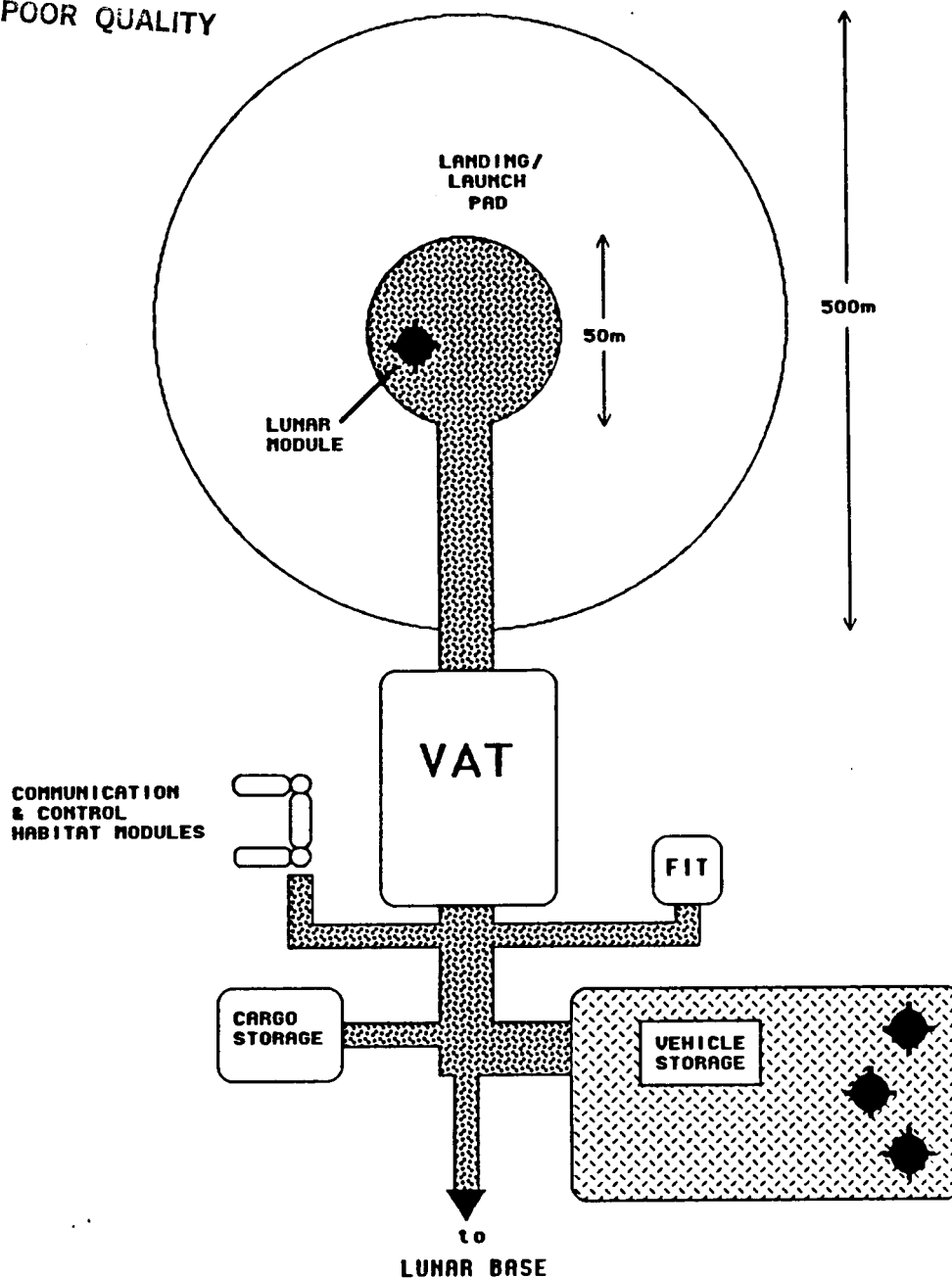


Figure 4: Complex 39L Plot Plan

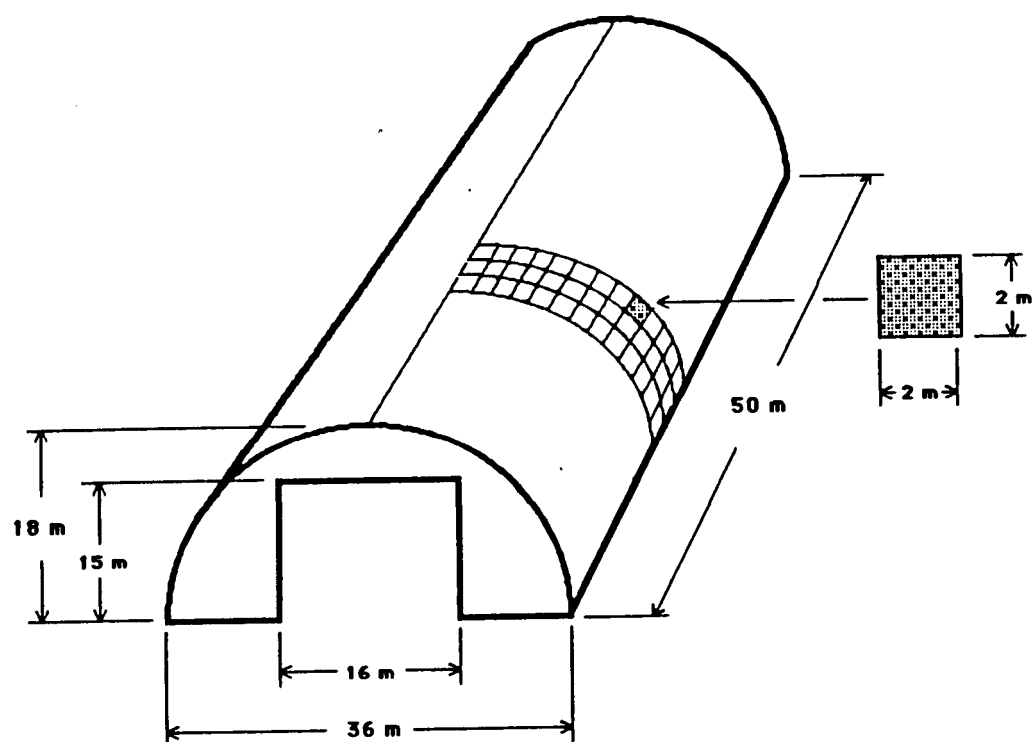


Figure 5: Complex 39L -- Vehicle Assembly Tent (VAT)

Figure 7: Lunar Landing and Launch Facility
(LLLF or Complex 39L) Design Matrix

	CONSTRUCTION				OPERATION				MAINTENANCE			
	SPECIFICATION	MASS	POWER	HANPOWER	SPECIFICATION	MASS	POWER	HANPOWER	SPECIFICATION	MASS	POWER	HANPOWER
LANDING / LAUNCH SITE CONSIDERATIONS												
SHELTER, STRUCTURE, SAFETY, ENVIRONMENTAL NEEDS												
LANDING / LAUNCH GUIDANCE, COMMUNICATIONS, COMPUTING NEEDS												
LUNAR MODULE SURFACE TRANSPORT SYSTEM												
HEAVY CARGO UNLOADING / LOADING SYSTEMS												
PERSONNEL UNLOADING / LOADING SYSTEMS												
PROPELLANT UNLOADING / LOADING SYSTEMS												
VEHICLE STORAGE												
MAINTENANCE, REPAIR, TEST & CHECK-OUT REQUIREMENTS												

APPENDIX 4

VECHICLE ASSEMBLY TENT

Preliminary Design

prepared for:

NASA/USRA/UADP

by:

Douglas W. Johnson
Mechanical Engineering Department
Florida Institute of Technology

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ABSTRACT

A preliminary design for a thermal shelter for the lunar landing module has been derived. The shelter consists of aluminized mylar panels covering an aluminum framework. A ground level temperature of less than 6 °C has been calculated. The total transferable mass of the structure is 1855 kg.

I. INTRODUCTION

A major design requirement of the lunar landing and launch facility, Complex 39L, is that it provide thermal shelter to the lunar modules being stored on the surface of the moon (ref. 1). A structure will be constructed large enough to contain four lunar modules, as well as the cargo crane and various module servicing equipment. The shelter will serve to protect the lunar module and other sensitive equipment from direct sunlight. Shading will decrease the boil-off of propellants from the module and also provide a more stable thermal environment, extending the useful life of the equipment stored inside. This report will examine a possible design for a lunar surface shelter.

Emphasis in design of the Vehicle Assembly Tent (VAT) will be placed on achieving the greatest possible shielding from the solar thermal spectrum while keeping the transferable weight of the structure to a minimum. The configuration of the VAT is as shown in figure 1. The semi-circular design will be 50 m long, 36 m wide and 18 m high at the peak. Reflective panels will be made of a mylar/aluminum film laminate that is expected to produce a reflectivity approaching 90%. The structural framework of the VAT will be constructed of 2014 aluminum.

II. DESIGN ANALYSIS

A. Thermal Analysis

The calculations presented in Appendix A show the one dimensional heat transfer analysis that was conducted. These calculations show that a lunar surface temperature on the order of 3 °C can be expected. The reflective panels utilized in this design are composed of a laminate of mylar for strength and evaporated aluminum film for reflectivity. The reflectivity of the aluminum film is 90% (ref. 2).

It will be assumed here that because the two panel configuration results in only a 9°C temperature difference as compared to the one panel scheme, that the extra mass and assembly time required of the two panel design will not be justified.

B. Structural Design

The structural design of the VAT consists of circular "ribs" located every two meters along the length of the tent. These ribs are each formed of 61 rods braced together at 3 degree angles to approximate a semicircle. The ribs are in turn braced together longitudinally. The constituents can be visualized in figures 2,3 and 4.

The rods are loaded mainly in compression and can therefore be considered columns. They are constructed of 2014-T6 aluminum due to its light weight and strength. The pertinent properties of this material are as follows (ref. 3):

Yield Strength, Compression	53000psi
Yield Strength, Shear	32000psi
Modulus of Elasticity	10900000psi
Density	2.70 g/cm

Appendix B gives a computer program that computes the minimum radius for these 1.8541 meter rods, given the weight of the reflective panels and connections. Using a factor of safety of 2, a theoretical radius of .11 cm was obtained. However, this gives a slenderness ratio (L/r) of $185.41/.11 = 1685.6$, which is beyond acceptable limits. A slight modification of the program to include the Euler equation for critical stress (ref. 4) yields a radius of .21, using the same factor of safety of two. We will increase our usable radius to .25 cm to compensate for end effects and material imperfections.

C. Reflective Panels

The reflective panels are composed of a laminate of mylar film and evaporated aluminum. The mylar has a thickness of 4 mils and a density of 1300 kg/m . The aluminum layer has a nominal thickness of 2 mils. This material combination has been utilized successfully in the past for passive thermal control of

satellites.

C. Weight Summary

The earth weight of each part of the shelter is as follows:

Weight of rod = $3.14159 \times .25 \times 185.41 \times 2.70 = 98.3$ grams

Weight of braces = $(3.14159 \times (.5 - .25) \times 20 + 30) \times 2.70$
= 112.8 grams

Weight of connectors = $3.14159 \times .1 \times 200 \times 2.70 = 17.0$ grams

Weight of mylar laminate =

weight of mylar = $2 \times 1.85 \times .000102 \times 1300 = 489.8$ grams

+weight of aluminum film = $2 \times 1.85 \times .000051 \times 2700 = 508.6$ grams
= 998.4 grams

The total weight of the structure is found by summing component weights:

Rods	1525	X	.0983	=	149.9	kg
Braces	1500	X	.1128	=	169.2	kg
Connectors	3750	X	.017	=	38.4	kg
Panels	1500	X	.9984	=	<u>1497.6</u>	<u>kg</u>
					1855.0	kg

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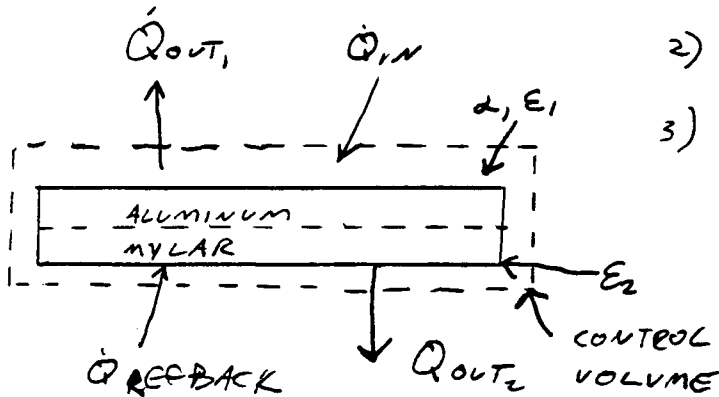
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Appendix A - Thermal Analysis

SINGLE SHIELD

ASSUMPTIONS:

- 1) THICKNESS OF PANEL NEGLIGIBLE
- 2) NO LONGITUDINAL CONDUCTION
- 3) STEADY STATE



$$\dot{E}_{IN} = \dot{E}_{OUT}$$

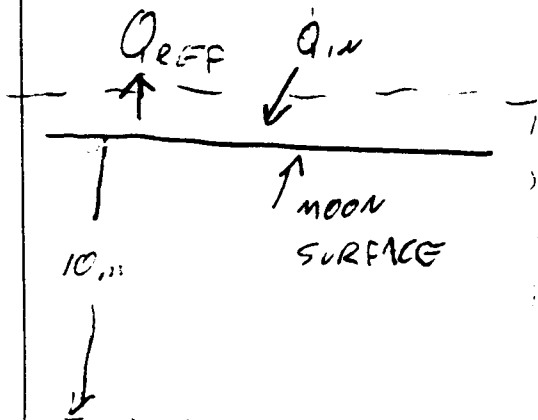
$$\dot{Q}_{IN} + \dot{Q}_{REFBACK} = \dot{Q}_{OUT1} + \dot{Q}_{OUT2}$$

$$\alpha G + (1 - \alpha_m) E_2 \sigma T_1^4 = E_1 \sigma T_1^4 + E_2 \sigma T_1^4$$

$$T_1 = \left(\frac{\alpha G}{\sigma (\alpha_m E_2 + E_1)} \right)^{1/4}$$

$$= \left(\frac{(61)(1353 \text{ W/m}^2)}{(5.67 \times 10^{-8} \frac{\text{W}}{\text{m}^2 \text{K}^4})((.4)(.8) + (.0))} \right)^{1/4}$$

$$= 283.4 \text{ K}$$



$$-20^\circ\text{C} = 253 \text{ K}$$

$$\dot{Q} = \frac{k \Delta T}{L}$$

$$\dot{Q} = \dot{Q}_{IN} - \dot{Q}_{REF}$$

$$\Delta T = T_s - T_{10m}$$

$$T_s = \frac{L \dot{Q}}{k} + T_{10m}$$

$$= \frac{L \alpha_m E_2 \sigma T_1^4}{k} + T_{10m}$$

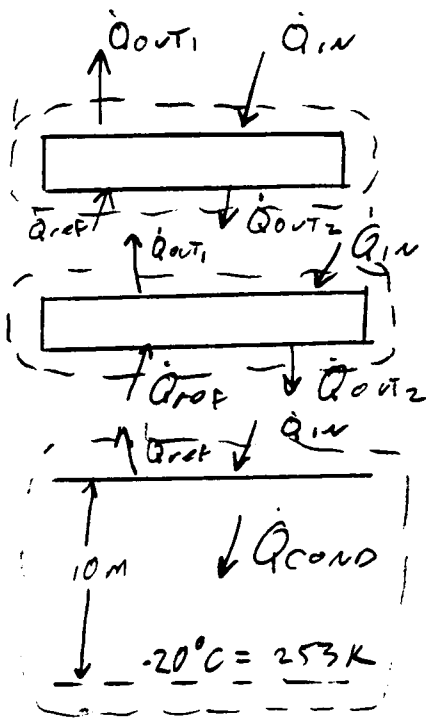
$$= \frac{(.4)(.8)(5.67 \times 10^{-8} \frac{W}{m^2 K^4})(10)(283K)^4}{(50 \frac{W}{m \cdot K})} + 253.2$$

$$= \underline{276.4 K}$$

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TANDEM SHIELDS

$$\dot{E}_{in} = \dot{E}_{out}$$

$$\dot{Q}_{in} + \dot{Q}_{ref} = \dot{Q}_{out1} + \dot{Q}_{out2}$$

$$\alpha G + (1-\alpha) \epsilon_2 \sigma T_1^4 = \epsilon_1 \sigma T_1^4 + \epsilon_2 \sigma T_1^4$$

$$T_1 = \left(\frac{\alpha G}{\sigma (\epsilon_2 + \epsilon_1)} \right)^{1/4}$$

$$= \left(\frac{(0.1)(1353 \text{ W/m}^2)}{(5.67 \times 10^{-8} \frac{\text{W}}{\text{m}^2 \text{K}^4})((0.1)(0.8) + 0.07)} \right)^{1/4}$$

$$= 368.1\text{K}$$

$$\dot{E}_{in} = \dot{E}_{out}$$

$$\dot{Q}_{in} + \dot{Q}_{ref} = \dot{Q}_{out1} + \dot{Q}_{out2}$$

$$\alpha \epsilon_2 \sigma T_1^4 + (1-\alpha_m) \epsilon_2 \sigma T_2^4$$

$$= \epsilon_1 \sigma T_2^4 + \epsilon_2 \sigma T_2^4$$

$$T_2 = \left(\frac{\alpha \epsilon_2 T_1^4}{\alpha_m \epsilon_2 + \epsilon_1} \right)^{1/4}$$

$$T_2 = \left(\frac{(0.1)(0.8)(368\text{K}^4)}{(0.4)(0.8) + 0.05} \right)^{1/4}$$

$$= 250.9$$

$$\dot{Q}_{cond} = \frac{k}{l} \Delta T = \dot{Q}_{in} - \dot{Q}_{ref}$$

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$$\Delta T = T_s - T_{10m}$$

$$T_s = \frac{lQ}{k} + T_{10m}$$

$$= \frac{l}{k} \alpha \epsilon_2 \sigma T_2^4 + T_{10m}$$

$$= \frac{10m}{50 \frac{W}{m \cdot K}} (.4)(.8) (5.67 \times 10^{-8} \frac{W}{m^2 K^4}) (250.9K)^4 + 253K$$

$$= 267.4 K$$

$\alpha =$ ABSORPTIVITY OF PANELS $= .1$
 $E_1 =$ EMISSIVITY OF PANELS $= .05$

$E_2 =$ EMISSIVITY OF MYLAR $= .8$

$\alpha_m =$ ABSORPTIVITY OF MOON $= .4$

$K =$ CONDUCTIVITY OF REGOLITH $= 50 \frac{W}{m \cdot K}$

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Appendix B - Structural Design Program

RUN	BEAM NUMBER	ANGLE FROM HORIZONTAL (DEGREES)	COMPRESSIVE FORCE (KG)	SHEAR FORCE (KG)
	1	0	22.20147	-.8353243
	2	3	22.33661	-.8200362
	3	6	22.57632	-.7938241
	4	9	22.92017	-.7567465
	5	12	23.36745	-.7088944
	6	15	23.91716	-.6503865
	7	18	24.56801	-.5813711
	8	21	25.31846	-.5020256
	9	24	26.16668	-.4125552
	10	27	27.11057	-.3131937
	11	30	28.14778	-.2042008
	12	33	29.27568	-8.586359E-02
	13	36	30.49143	.0415057
	14	39	31.79191	.1775701
	15	42	33.1738	.3219686
	16	45	34.63354	.4743177
	17	48	36.16736	.634212
	18	51	37.77128	.8012253
	19	54	39.44115	.9749121
	20	57	41.17262	1.154808
	21	60	42.96118	1.340433
	22	63	44.80215	1.53129
	23	66	46.69072	1.726867
	24	69	48.62195	1.926642
	25	72	50.59079	2.130079
	26	75	52.59207	2.336633
	27	78	54.62054	2.545749
	28	81	56.67087	2.756867
	29	84	58.73768	2.969421
	30	87	60.81555	3.18284
	31	90	62.89901	3.396552

MINIMUM RADIUS OF MEMBERS (CM)

.11

OK

EARTH WEIGHT OF MEMBERS (KG)

1.902969E-04

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BEAM NUMBER	ANGLE FROM HORIZONTAL (DEGREES)	COMPRESSIVE FORCE (KG)	SHEAR FORCE (KG)
1	0	42.38466	.2209814
2	3	42.5475	.2377191
3	6	42.81494	.2653827
4	9	43.18657	.3039137
5	12	43.66166	.3532217
6	15	44.23921	.413187
7	18	44.91796	.483662
8	21	45.69634	.5644692
9	24	46.57252	.6554031
10	27	47.54441	.7562303
11	30	48.60966	.8666908
12	33	49.76565	.9864975
13	36	51.00951	1.115339
14	39	52.33815	1.252877
15	42	53.74824	1.398751
16	45	55.23621	1.552578
17	48	56.7983	1.713952
18	51	58.43054	1.882447
19	54	60.12876	2.057617
20	57	61.88862	2.238999
21	60	63.7056	2.426112
22	63	65.57505	2.618459
23	66	67.49213	2.815529
24	69	69.45191	3.016798
25	72	71.44933	3.221731
26	75	73.47924	3.429783
27	78	75.53638	3.6404
28	81	77.61542	3.85302
29	84	79.71098	4.067078
30	87	81.81763	4.282004
31	90	83.92991	4.497224

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MINIMUM RADIUS OF MEMBERS (CM)

EARTH WEIGHT OF MEMBERS (KG)

21 6.935618E-04

```

10 REM*****
20 REM
30 REM  THIS BASIC PROGRAM WILL DERIVE A SIZE FOR THE SUPPORT MEMBERS OF THE VAT
40 REM      AND WILL CALCULATE THE FORCES THAT ARE PRESENT THEREIN
50 REM
60 REM  VARIABLES:
70 REM      FS=SHEAR FORCE
80 REM      FR=RESULTANT FORCE (COMPRESSIVE)
90 REM      L=LENGTH OF BEAMS (METERS)
100 REM      ERHO=DENSITY OF ALUMINUM (KG/CM3)
110 REM      WM=MOON WEIGHT OF MYLAR (KG)
120 REM      WC=MOON WEIGHT OF CONNECTIONS (KG)
130 REM      R=RADIUS OF BEAMS (CM)
140 REM      THETA=RADIUS OF BEAM FROM HORIZONTAL (DEGREES)
150 REM      RHETA=THETA IN RADIANS
160 REM      MOM=MOMENT PRODUCED BY WEIGHT OF STRUCTURE (KG*M)
170 REM      FRMAX=MAXIMUM COMPRESSIVE FORCE (KG)
180 REM      PG=MAXIMUM COMPRESSIVE STRESS(KG/CM3)
190 REM
200 REM*****
210   DIM FS(35)
220   DIM FR(35)
230   LET PI=3.14159
240   LET L=1.8541
250   LET ERHO=.0027
260   LET MRHO=ERHO/6
270   LET WM=1
280   LET WC=1
290   LET R=.01
300   LET WB=PI*R*R*(L/100)*MRHO+WM+WC
310   LET THETA=0
320   WHILE THETA < 90
330     LET RHETA=THETA/57.29578
340     LET MOM=MOM+WB*L/2*COS(RHETA)
350     LET THETA=THETA+3
360   WEND
370   LET FR(1)=MOM/18
380   LET X=2
390   LET THETA=3
400   WHILE X < 33
410     LET RHETA=3/57.29578
420     LET GAMMA=THETA/57.29578
430     LET FR(X)=FR(X-1)/COS(RHETA)+WB*SIN(GAMMA)
440     LET FS(X-1)=FR(X-1)*SIN(RHETA)-WB*COS(GAMMA)
450     LET X=X+1
460     LET THETA=THETA+3
470   WEND
480   LET P=1
490   WHILE P < 32
500     LET Q=P+1
510     IF FR(P) > FR(P-1) THEN FRMAX=FR(P)
520     ELSE FRMAX=FR(P-1)
530     LET P=P+1
540   WEND
550   LET PG=(2*FRMAX)/(PI*R*R)
560   IF PG < 3734.098 THEN GOTO 590
570   LET R=R+.01
580   GOTO 300
590   PRINT TAB(4); "BEAM"; TAB(13); "ANGLE FROM "; TAB(27); "COMPRESSIVE"; TAB(47); "SHE
AR"
600   PRINT TAB(3); "NUMBER"; TAB(13); "HORIZONTAL"; TAB(30); "FORCE"; TAB(47); "FORCE"
610   PRINT TAB(14); "(DEGREES)"; TAB(31); "(KG)"; TAB(49); "(KG)"

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```
620 PRINT
630 LET THETA=0
640 LET S=1
650 WHILE S < 32
660   PRINT TAB(4);S;TAB(16);THETA;TAB(28);FR(S);TAB(44);FS(S)
670   LET S=S+1
680   LET THETA=THETA+3
690 WEND
700 PRINT
710 PRINT
720 PRINT "MINIMUM RADIUS OF MEMBERS (CM)";TAB(40);"EARTH WEIGHT OF MEMBERS (KG)
"
730 PRINT
740 PRINT R;TAB(40);PI*R*R*L*ERHO
750 END
```

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```

10 REM*****
20 REM
30 REM THIS BASIC PROGRAM WILL DERIVE A SIZE FOR THE SUPPORT MEMBERS OF THE VAT
40 REM AND WILL CALCULATE THE FORCES THAT ARE PRESENT THEREIN
50 REM
60 REM VARIABLES:
70 REM FS=SHEAR FORCE
80 REM FR=RESULTANT FORCE (COMPRESSIVE)
90 REM L=LENGTH OF BEAMS (METERS)
100 REM EHRO=DENSITY OF ALUMINUM (KG/CM3)
110 REM WM=MOON WEIGHT OF MYLAR (KG)
120 REM WC=MOON WEIGHT OF CONNECTIONS (KG)
130 REM R=RADIUS OF BEAMS (CM)
140 REM THETA=RADIUS OF BEAM FROM HORIZONTAL (DEGREES)
150 REM RHETA=THETA IN RADIANS
160 REM MOM=MOMENT PRODUCED BY WEIGHT OF STRUCTURE (KG*M)
170 REM FRMAX=MAXIMUM COMPRESSIVE FORCE (KG)
180 REM PG=MAXIMUM COMPRESSIVE STRESS(KG/CM3)
190 REM
200 REM*****
210 DIM FS(35)
220 DIM FR(35)
230 LET PI=3.14159
240 LET L=1.8541
250 LET ERHO=.0027
260 LET MRHO=ERHO/6
270 LET WM=.25
275 LET K=.875
280 LET WC=1
290 LET R=.01
300 LET WB=PI*R*R*(L/100)*MRHO+WM+WC
310 LET THETA=0
320 WHILE THETA < 90
330 LET RHETA=THETA/57.29578
340 LET MOM=MOM+WB*L/2*COS(RHETA)
350 LET THETA=THETA+3
360 WEND
370 LET FR(1)=MOM/18
380 LET X=2
390 LET THETA=3
400 WHILE X < 33
410 LET RHETA=3/57.29578
420 LET GAMMA=THETA/57.29578
430 LET FR(X)=FR(X-1)/COS(RHETA)+WB*SIN(GAMMA)
440 LET FS(X-1)=FR(X-1)*SIN(RHETA)-WB*COS(GAMMA)
450 LET X=X+1
460 LET THETA=THETA+3
470 WEND
480 LET P=1
490 WHILE P < 32
500 LET Q=P+1
510 IF FR(P) > FR(P-1) THEN FRMAX=FR(P)
520 ELSE FRMAX=FR(P-1)
530 LET P=P+1
540 WEND
550 LET PG=(2*FRMAX)/(PI*R*R)
551 LET FA= 53789344#/(K*(L*100)/R)^2)
560 IF PG < 3734.098 AND FRMAX < FA THEN GOTO 590
570 LET R=R+.01
580 GOTO 300
590 PRINT TAB(4); "BEAM"; TAB(13); "ANGLE FROM "; TAB(27); "COMPRESSIVE"; TAB(47); "SHE
AR"

```

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610 PRINT TAB(14); "(DEGREES)"; TAB(31); "(KG)"; TAB(48); "(KG)"
620 PRINT
630 LET THETA=0
640 LET S=1
650 WHILE S < 32
660 PRINT TAB(4); S; TAB(16); THETA; TAB(28); FR(S); TAB(44); FS(S)
670 LET S=S+1
680 LET THETA=THETA+3
690 WEND
700 PRINT
710 PRINT
720 PRINT "MINIMUM RADIUS OF MEMBERS (CM)"; TAB(40); "EARTH WEIGHT OF MEMBERS (KG)"
730 PRINT
740 PRINT R; TAB(40); $\pi * R * R * L * ERHO$
750 END

070

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APPENDIX 5

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FINAL REPORT

MAINTENANCE AND SAFETY CONCERNS FOR A LUNAR LAUNCH AND LANDING FACILITY

NADINE GEIER
DEPARTMENT OF ELECTRICAL AND COMPUTER ENGINEERING
FLORIDA INSTITUTE OF TECHNOLOGY
MELBOURNE, FL

JUNE 6, 1988

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ABSTRACT

A definition of the general safety and maintenance requirements for a lunar launch and landing facility has been formulated. For this report, a permanently-manned lunar base, a Low Earth Orbit (LEO) space station, a Low Lunar Orbit (LLO) space station, and a multipurpose, reusable lunar module are assumed to already exist. Six areas of concern covered in this report are Range Safety, Emission Particle Damage, Radiation Exposure Damage, Maintenance, Turnaround Schedule, and Computer Systems.

OUTLINE

RANGE SAFETY

EMISSION PARTICLE DAMAGE

RADIATION EXPOSURE DAMAGE

MAINTENANCE

TURNAROUND SCHEDULE

COMPUTER SYSTEMS

TABLES

TABLE 1: LANDING BLAST EJECTA ON THE MOON

TABLE 2: ANNUAL DOSE EQUIVALENT DUE TO COSMIC-RAY GENERATED
NEUTRONS

FIGURES

FIGURE 1: EARTH -- MOON TRANSPORTATION INFRASTRUCTURE

FIGURE 2: COMPLEX 39L

FIGURE 3: HABITAT MODULES

FIGURE 4: SHIELDING REQUIREMENTS

FIGURE 5: VEHICLE ASSEMBLY TENT (VAT)

FIGURE 6: DOSE DUE TO COSMIC-RAY NUCLEI WITH SHIELDING

FIGURE 7: SOLAR FLARE MODELS

FIGURE 8: SECONDARY NEUTRON GENERATION

FIGURE 9: DOSE FROM SECONDARY NEUTRONS AND COSMIC-RAY
NUCLEI

FIGURE 10: LUNAR MODULE TURNAROUND SCHEDULE

INTRODUCTION

A general description of the safety and maintenance considerations for a lunar launch and landing facility has been defined. We assume a phase III lunar base with enough power available to meet the demands of a launch and landing facility. We also assume multipurpose, reusable lunar vehicles will exist that will have a flight schedule of 6, 12, or 24 flights/year. The specific design of the lunar base and the lunar vehicle are beyond the scope of this paper.

The range safety and abort modes for ascents and descents are explained in the first section of this report. Then possible damage caused by particles thrown from the pad during launch and landing activities are studied. Next, the possibility of human and electronic damage caused by radiation is shown. Regular maintenance and repair scenarios on the lunar lander is looked at in the fourth section. Then the turnaround schedule from landing to launch is defined. Finally, the computer systems needed for the Launch Processing System are briefly described.

RANGE SAFETY

In studying any launch and landing facility, whether it be on the Earth or on the Moon, abort modes and range safety must be considered (Ref. 14). In all cases where an abort might be necessary, the safety of the personnel must be the prime consideration. If the lunar lander is unmanned, the safety of the ground personnel and facilities would take precedent over that of the lunar module. If, however, the lunar lander is manned, a tradeoff might occur between the safety of the of the ground personnel and facilities and that of the module personnel and equipment. To reduce the risk of human injury in both cases, all ground personnel should be in protected environments during launch and landing operations.

During the ascent phase, there are three possible abort modes: on-pad aborts, abort-to-surface, and abort-to-orbit. In all these abort modes, the primary response of the base should be emergency crew removal. An on-pad abort would be the result of a critical failure occurring before liftoff from the pad area. For this mode, the feasibility of on-pad emergency evacuation equipment should be studied.

An abort-to-surface would be the result of a critical failure occurring during liftoff from the pad. In most cases, the crew could be reasonably expected to land the module safely on the surface, with minimal or no damage to the lander and without damage to base facilities. Of course, if the module lands downrange, a rescue team should be immediately dispatched to the site. If the capsule becomes uncontrollable and heads toward a populated base section, consideration must be given to a self-destruct option.

An abort-to-orbit would be the result of a critical failure occurring during liftoff from the pad but after enough velocity has been obtained to secure an orbit around the Moon. In this case, the crew must try to maneuver the vehicle into a link with the space station (Figure 1). In all likelihood, this link should be possible. If it is not possible, the module's orbit around the Moon should decay and a controlled landing would be attempted.

During the descent phase, there are also three possible abort modes: an abort-to-orbit, abort-to-surface, and bad landings. Again, emergency crew removal should be the base's primary goal. An abort-to-orbit would be the result of a critical failure occurring almost immediately after the module is released from the space station. The options of linking with the space station or a controlled landing on the surface apply in the decent phase as well as the ascent phase.

An abort-to-surface would be the result of a critical failure occurring at a point where it would be impossible for the module to return to orbit. The same options apply for descent phase that apply for the ascent phase. In either the ascent or descent phase of operation, the computer systems should be able to decide whether the abort-to-surface mode or the abort-to-orbit mode is more appropriate.

A bad landing would be the result of a critical failure occurring at the final phases of touchdown on the pad. In most cases, a bad landing is not dangerous or extremely harmful to the module. However, precautions should be taken just in case.

EMISSION PARTICLE DAMAGE

The effects of rocket engine blast on both paved and unpaved pads must be considered in the analysis of any launch and landing facility on the Moon (Ref. 14). As the result of engine blast, not only paved pads would be damaged, but neighboring facilities and equipment as well. Of course, the damage caused by launch and landing activities on a paved pad would be considerably less to surrounding surface equipment and facilities than that of an unpaved pad. However, the possibility of particles small enough to be ejected still exists with a paved pad. The same sort of precautions, therefore, must be used for paved and unpaved pads.

There are many reasons that the launch and landing facilities must take particle damage caused by engine blast into account. Coverings must be designed to protect the more sensitive equipment from contamination caused by dust kicked up from engine blast. The distance between the pad and the surface facilities and equipment will depend on how far away blast damage can occur. The design and protection of equipment that must remain in the vicinity of the pad will be governed by how serious this damage will be. In addition, the design of the permanent paved pad itself will be effected by concern over engine blast damage.

However, even though there will be engine blast damage and range safety considerations, support facilities should be located as close to the launch and landing pad as feasible. This will allow time and risks for module maintenance and crew transfer to be reduced. Therefore, protective covers and/or protective walls should be seriously considered to provide protection for surface equipment and facilities. In addition, personnel within 3 to 5 km of the pad should be behind a protective surface during any launch or landing activity.

The relative size and velocity of typical particles ejected by engine blast can be seen in Table 1. In general, the particles ejected by blast will be low velocity, relative to typical meteoroid velocity. The maximum ejected particle is expected to be approximately 5 millimeters, traveling at 10 meters per second. Logically, smaller particles will have higher velocities and will, thus, travel farther.

The relative effects of engine blasts can be seen in a description of the damage occurred by surfaces at various distances. The following is an excerpt from the report Lunar Base Launch and Landing Facility Conceptual Design by Eagle Engineering (March 1988) and applies to unpaved pads.

0 to 50 meters

Metal objects will experience significant surface damage after only one landing and glass surfaces will experience severe damage after one landing. Vision glass will be virtually unusable after one landing.

50 through 200 to 400 meters

Metal objects will experience significant pitting damage after several landings but only minor pitting after one landing. Glass surfaces on the other hand will experience significant damage after one landing and be unusable after several landings.

400 meters to past 2 kilometers

Metal objects will sustain only very minor and probably unnoticeable pitting damage after numerous landings. Reflective surfaces should be protected. Glass objects will sustain minor damage after numerous landings. The damage will eventually be unacceptable for optical quality glasses. Optical instruments should face away from landings.

RADIATION EXPOSURE DAMAGE

The Fuel Inventory Tent (FIT), the Vehicle Assembly Tent (VAT), the Vehicle Storage area, and the Cargo Storage area are all places in which people and equipment will be located. Although these structures are designed to control extreme temperature fluctuations, they were not designed to block out radiation. Therefore, when personnel and equipment are inside these tents, they are subject to radiation exposure like that on the lunar surface.

On Earth, radiation exposure is a fairly common experience. On the average, an adult in the United States receives 100 mrem/year from cosmic radiation and naturally occurring radioactive material (Ref. 16). The established annual maximum dosage for the average adult is 0.5 rem/year. Fortunately, unlike the Moon, the Earth is protected from most radiation by its magnetic field and thick atmosphere.

The Moon has no magnetic field or atmosphere to protect its surface from radiation. Lunar surface radiation is 30 rem/year during normal solar activity. This increased level of radiation could effect humans and electrical equipment adversely. On the Earth, materials such as lead and concrete are used to protect people and equipment from high radiation levels. Unfortunately, these materials are not found naturally on the Moon and are too heavy to be economically feasible to ship. Burying modules in the lunar soil, or regolith, is a popular solution to combat radiation exposure.

There are two main types of radiation. One type is Galactic Cosmic Rays (GCR), see Figure 6 and Table 2. These rays are the remnant radiations from the Big Bang and are constantly striking the lunar surface. The second type is from the Sun. Solar Energy Particles (SEP) are the elements that are released from the Sun during a solar flare or are particles energized in the Sun's vicinity, see Figure 7. During solar flares, or SEP events, the annual equivalent dose can exceed 1000 rem in a matter of hours. GCR's are considered low-level radiation which human beings can absorb for limited times without adverse effects on the body. SEP's, on the other hand, can seriously and irreversibly harm a non-protected human.

Since personnel will have to work on the Moon's surface sometimes, the maximum radiation dosage was raised to that of an Earth-based nuclear facility worker, 5 rem/year (Ref. 4). With this standard, personnel can spend 20% of their time on the lunar surface with little or no protection (Ref. 9), like in the VAT, FIT, Vehicle Storage area, or Cargo Storage area. In this allotted time, the workers could perform maintenance, load/unload cargo, store propellants, etc. During solar flares, it will not be safe to be anywhere but in a protective bunker.

When the lunar workers are not on the surface or in the tents, they will be in modules buried under at least 2.5 meters (400 g/cm^2) of densely packed lunar soil. During SEP events, all of the personnel will have to be in protective bunkers that are at least 4 meters (700 g/cm^2) under densely packed lunar soil (Ref. 9).

Unfortunately, burying the modules under regolith poses some additional problems. As the regolith density increases, the nuclei present in the soil are energized by the incident GCR/SEP nuclei. These energized nuclei then, in turn, energize still more nuclei. This chain reaction, seen in Figure 8, could clearly be detrimental to any human beings or electronic equipment located beneath the regolith shield. In fact, the annual dose equivalent due to the secondary neutron generation actually increases to about 2 times the acceptable level at approximately 1 meter below the regolith shielding, see Figure 9. Obviously, more research is required into secondary neutron generation and regolith shielding.

Lunar workers are not the only things effected by radiation exposure. Electronic devices are also effected by their total accumulated radiation dosage from GCR, SEP, and secondary neutrons (Ref. 4). The resulting problems are called Hardware Upsets because they cause electrical hardware errors. The severity of these problems can be reduced by implementing three preventive steps. One, use only components that have been specifically designed to tolerate large doses of radiation. Two, store as much equipment as possible in protected environments. Three, perform regular systems checks on equipment, both visually and electronically.

Electronic components can also be effected by single, intensely ionizing particles like those found in SEP events (Ref. 4). The resulting problems are called Software Upsets because they cause unexpected software changes in programs. The severity of these problems can again be reduced by implementing three preventive steps. One, use redundant software programs. Two, use programs that have fault tolerance. Three, regularly check the software against identical, accurate software and correct any errors. Electronic components have already been developed that take Hardware and Software Upsets into account. In the future, however, electrical systems should be able to deal with radiation exposure more effectively.

MAINTENANCE

The lunar module is a reusable vehicle and will require regular maintenance with each flight. Vehicle maintenance will include but is not limited to: vehicle test and check-out, lubrication, recharging of environmental systems, recharging or replacing batteries, regenerating fuel cells, modular parts replacement/repair, and system modifications. Unlike presently operated reusable terrestrial space vehicles, the lunar capsule should require minimal maintenance. For our highest frequency flight schedule (24 flights/year), the lunar module turnaround time will be two weeks (14 days). For a baseline case we assume that routine maintenance will be performed by two personnel. However, more manpower will be required if a significant problem develops, if major system alterations are required, or if the number of capsules exceeds manpower capability.

The lunar capsule should be modular by design so that components can be replaced or removed with ease by personnel wearing space suits. If a lunar module component is damaged it will be replaced by a new component or one from another vehicle. If the component is beyond repair, it will be discarded and a new component will be installed. In some cases, when the component will be repaired at a later date, it will be replaced and removed to a repair storage area until it can be fixed.

When a lunar module component malfunctions or is damaged, it can either be repaired or "scrapped". Repairs can be performed on-site, at one of the space stations, or on earth. In many cases it will save time and money to repair the component at Complex 39L. Lunar module mechanics will be constrained by space suits making detailed repairs difficult if not impossible. Smaller components (computer systems, circuit boards, etc.) can be brought into the Habitat Modules, see Figure 3, to be worked on in a shirt-sleeve environment. Detailed repairs on larger components (engines, fuel tanks, etc.) will require a large pressurized facility. We believe that such a facility is beyond a Phase III Lunar Base, and repairs of this nature will not be possible at Complex 39L at this time. We assume that there will be tasks, such as replacing bolts and rivets, that can be performed adequately by personnel in space suits. Specially designed tools and equipment will be required.

In some cases, the repairs will be too extensive and the vehicle will be labeled unusable. If this is the case the vehicle will be towed to vehicle storage as previously described and a new vehicle sent from earth to replace it, if necessary. Working components on the stored lunar module will be cataloged for use in the repair of operational vehicles. Spare parts equivalent to at least one lunar module should be stored on hand at Complex 39L (Ref. 7).

TURNAROUND SCHEDULE

The lunar capsule is a reusable vehicle that will require inspection and maintenance after each flight. As previously stated, the typical turnaround time for a two person maintenance crew is 14 days. There are four main procedures involved in the typical turnaround schedule (Ref. 11). They include: Initial Safing, Postflight Servicing, Lunar Module Modifications, and Preflight Servicing. The following is a preliminary description of some of the operations that will be performed during each of our identified main procedures.

Initial Safing will include: transportation of the lunar module into the VAT, unloading and draining of fuel tanks, unloading and safing payloads attachment of ground power and purge lines to the lunar module, purging of main engines and fuel lines to remove possible moisture resulting from hydrogen/oxygen combustion, and removing waste products. Also, the lunar module crew will disembark sometime during the initial safing procedure. This is a preliminary list of required operations, that can easily be expanded upon.

After Initial Safing is complete, Postflight Reservicing begins. Postflight Reservicing will include: troubleshooting to determine anomalies which may have occurred during launch, spaceflight, or landing, visual and electrical inspections, and routine servicing. An umbilical containing electrical, communication, instrumentation, and control lines is connected to the vehicle to aid in troubleshooting the lunar module. Along with postflight inspection, routine servicing will include: lubrication, recharging of environmental systems, recharging or replacing batteries, regenerating fuel cells, and other required routine maintenance tasks.

Lunar Module Modifications can be made any time after Initial Safing is complete and before functional checks are made during Preflight Servicing, if necessary. Modifications could include: replacing damaged components that were discovered during the postflight inspection, adding or removing equipment necessary to meet future mission requirements, and replacement of outdated hardware with new designs to enhance vehicle performance. Lander modifications, if extensive, can be performed over a long period of time while the craft is in vehicle storage. However, many modifications will be performed in parallel with routine servicing.

The lunar module will then be prepared for launch with Preflight Servicing. This will include: installation of flight supplies and payload, attachment of fuel tanks, loading of personnel, visual and electronic functional checks of lunar module systems, detachment of ground umbilicals, and transportation of the vehicle from the VAT to the launch pad. Any system that fails the functional tests will undergo troubleshooting to identify the problem. If required, subsequent repairs or replacements are performed. Again, many more operations can be included.

For the four main procedures we develop a preliminary Lunar Module Turnaround Schedule (Figure 10). This figure was derived from preliminary estimations for the time requirements of the four main procedures. We recognize that this, though an overly simplistic representation, is the most detailed definition of reusable lunar module servicing to date.

COMPUTER SYSTEMS

To accomplish all the tasks required in the allotted time frame, a sophisticated computer system, similar to the Space Shuttles', will be needed (Ref.11). We assume that the technology will exist to implement the desired Launch Processing System required by the lunar launch and landing facility. We also assume that the system will be mostly automated to free the maintenance personnel to work on other areas.

The Launch Processing System (LPS) controls all the computer systems involved with the launch and landing aspects of the module. Specifically, the LPS controls and performs much of the vehicle check-out automatically while the vehicle components are being prepared for launch. It also conducts the countdown and launch operations and provides the capability for work order control and scheduling.

The Launch Processing System is divided into three major subsystems: Central Data Subsystem; Check-out, Control, and Monitor Subsystem; and Record and Playback Subsystem. The Central Data Subsystem stores test procedures, vehicle processing data, master program library, and pre/post flight test data analysis. This subsystem consists of large scale computers.

The Check-out, Control, and Monitor Subsystem performs vehicle check-out, countdown, landing, and launch. Pre-determined measurements related to test requirements, launch commit criteria, and performance specifications are stored in these computers. When the check-out program is complete, a signal indicates whether or not its performance has been satisfactory. If unsatisfactory, the computer will then provide data which will help isolate the fault. This subsystem consists of consoles, minicomputers, and other related equipment.

The Record and Playback Subsystem records lander instrumentation data during tests, landings, and launch countdowns. These recordings then can be played back for analysis when troubleshooting lander module anomalies. This subsystem consists of instrumentation tape recorders, telemetry demultiplexing equipment, direct-write recorders, and computers to provide data reduction capabilities.

PARTICLE DIAMETER (MM)	IMPACT DISTANCE (M)	IMPACT VELOCITY (M/SEC)
4.00	20	10
2.00	40	15
1.50	50	20
1.00	75	25
0.50	150	35
0.25	325	50
0.075	1200	100
0.050	2000	125

TABLE 1: LANDING BLAST EJECTA ON THE MOON (Ref. 14)

DEPTH (G/CM ²)	ANNUAL DOSE (REM/YEAR)
0	1.5
10	3.0
20	5.0
100	13.0
200	12.0
300	8.0
400	5.0
500	2.0

TABLE 2: ANNUAL DOSE EQUIVALENT DUE TO COSMIC-RAY GENERATED NEUTRONS (Ref. 8)

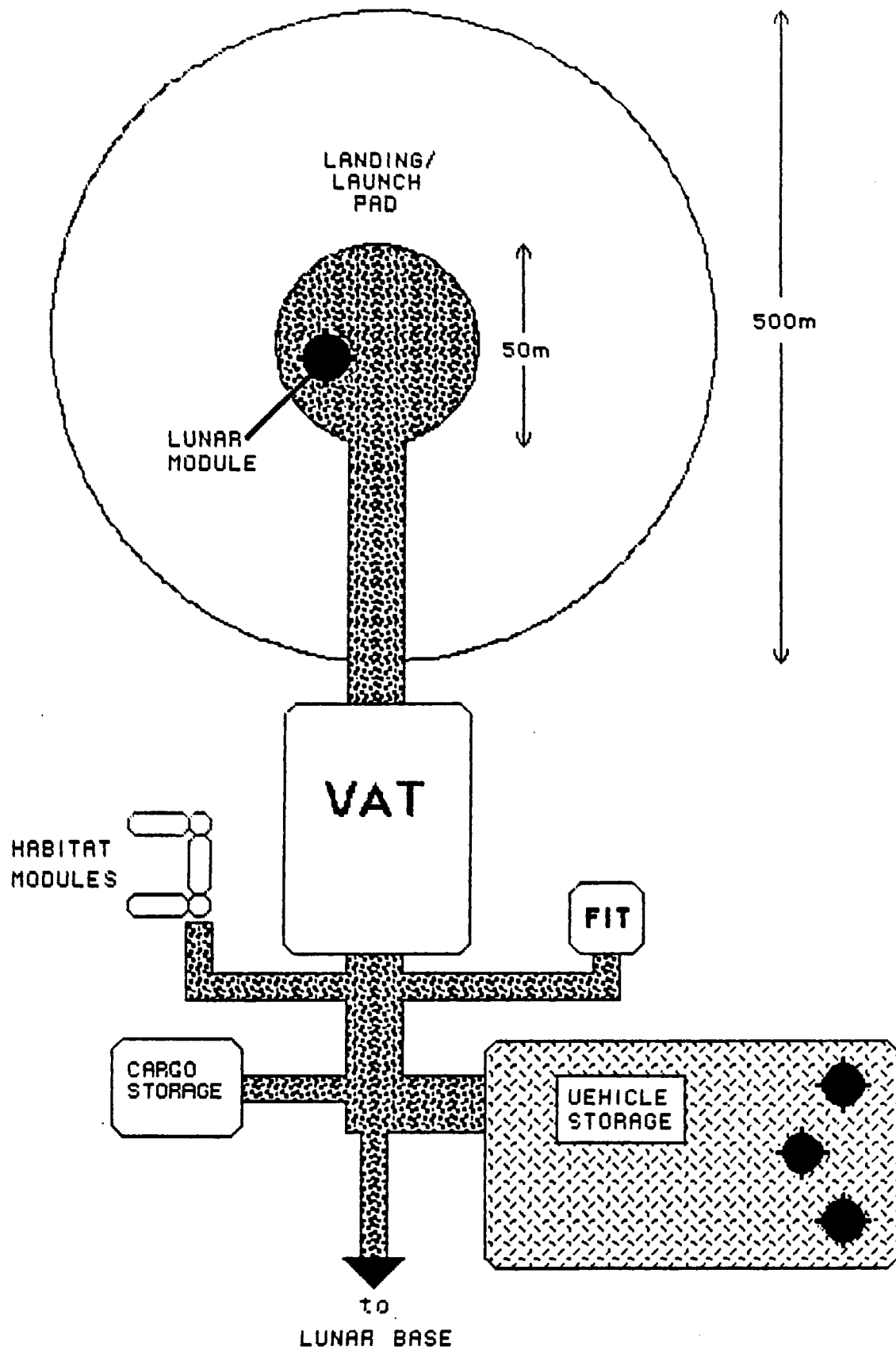


FIGURE 2: COMPLEX 39L

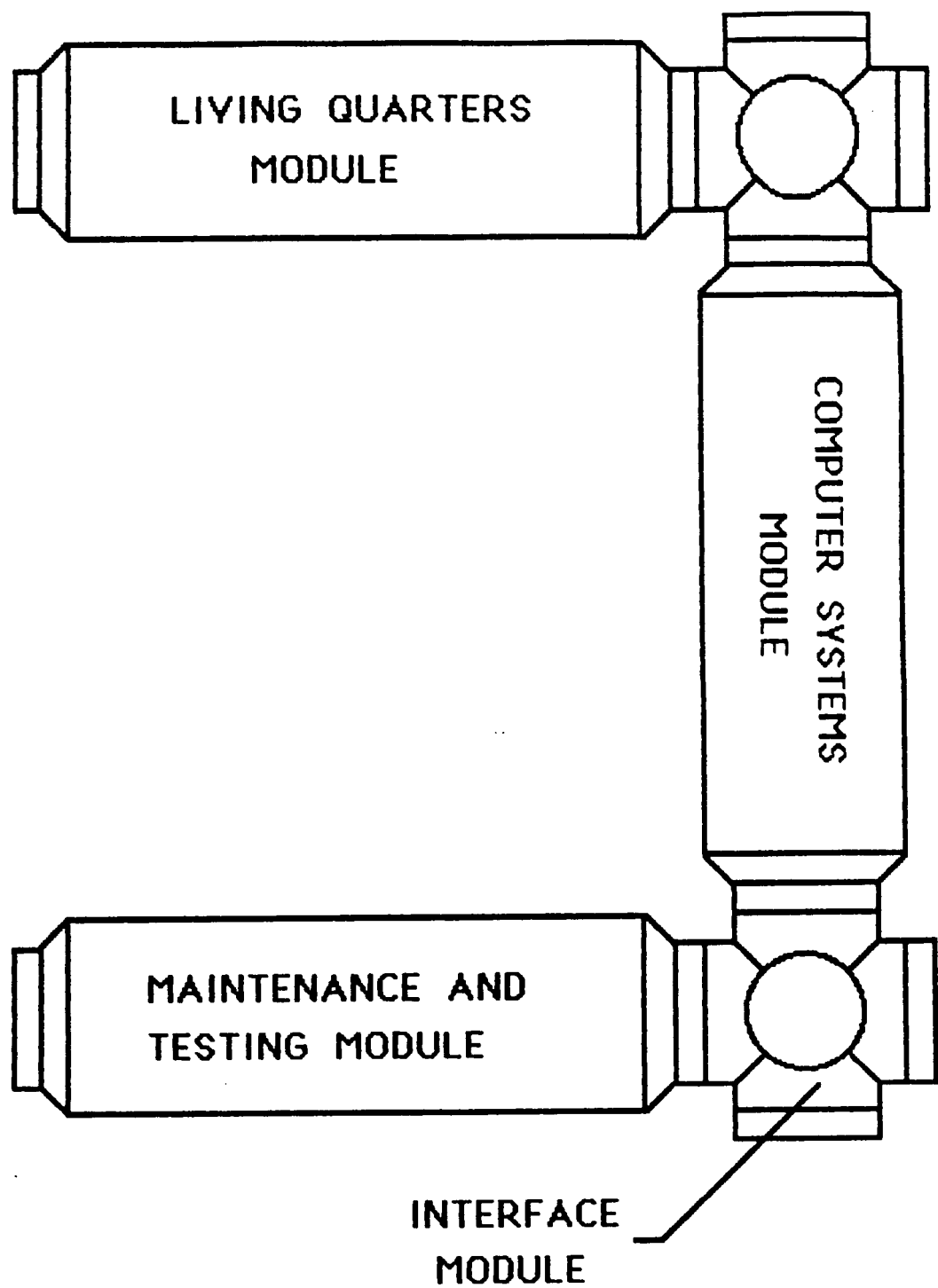


FIGURE 3: HABITAT MODULES (Ref. 12)

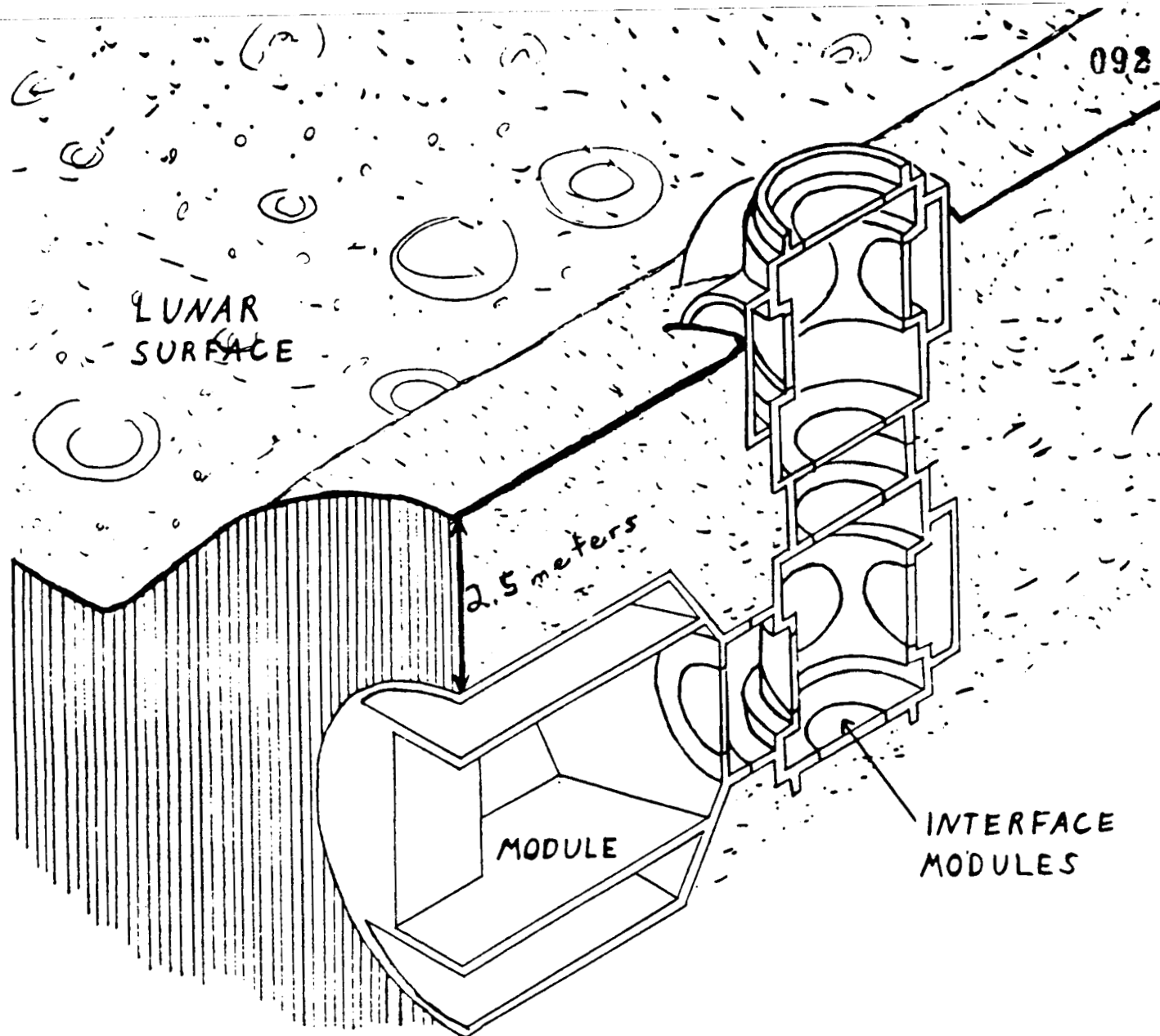


FIGURE 4: SHIELDING REQUIREMENTS (Ref.12)

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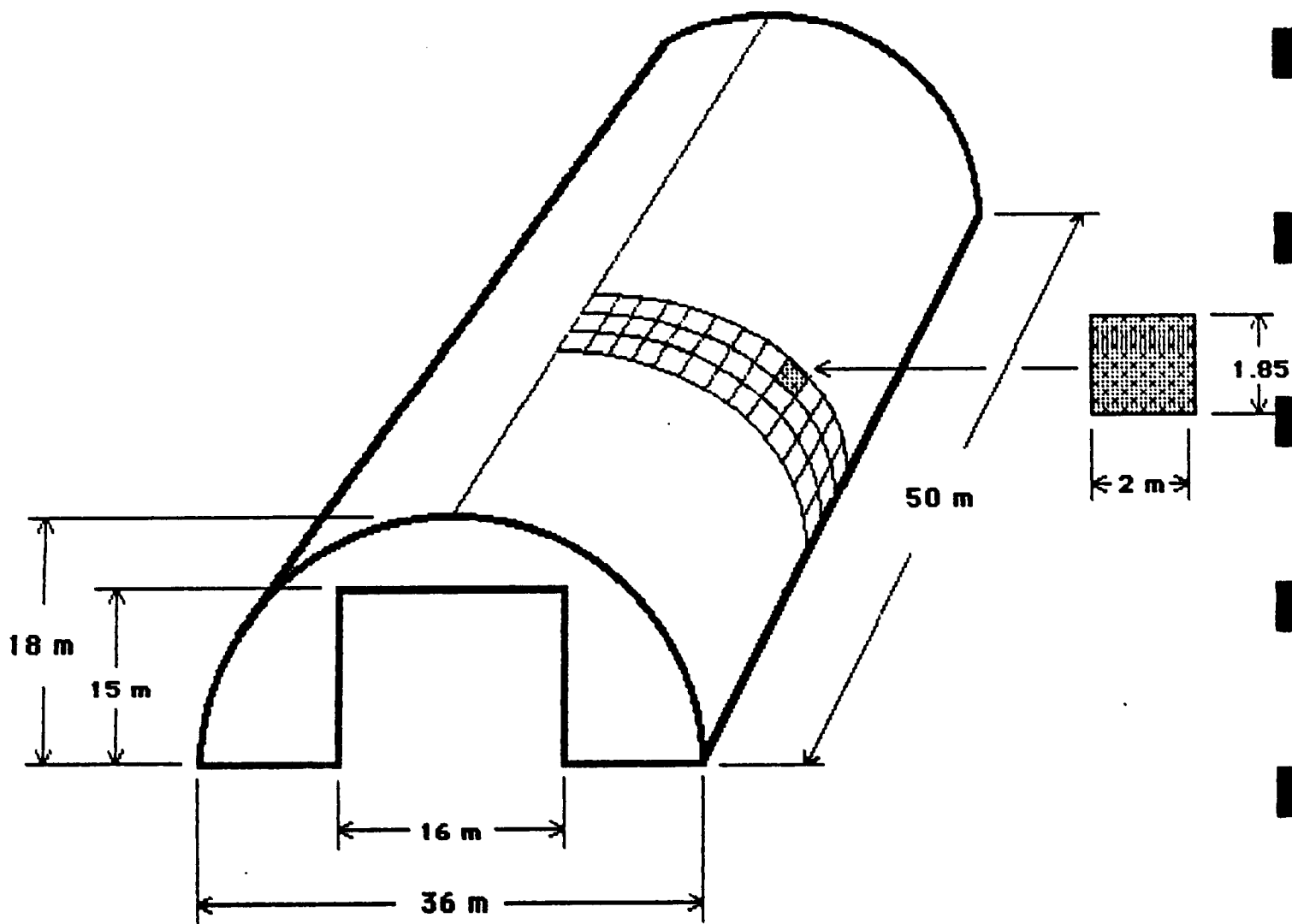
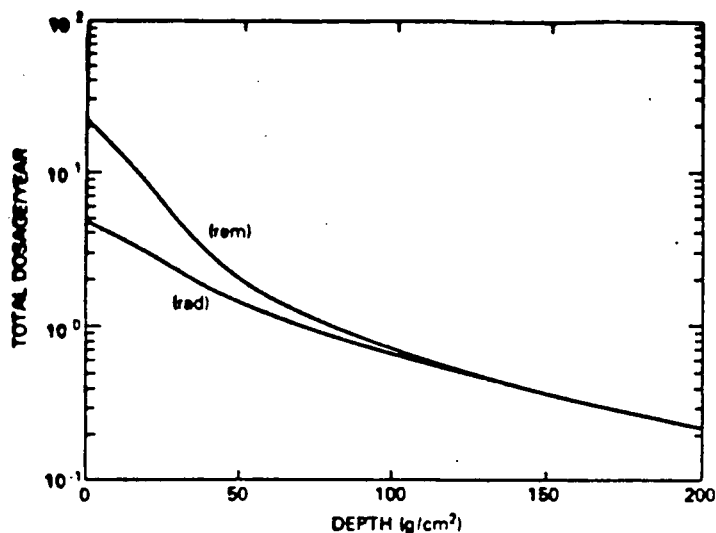


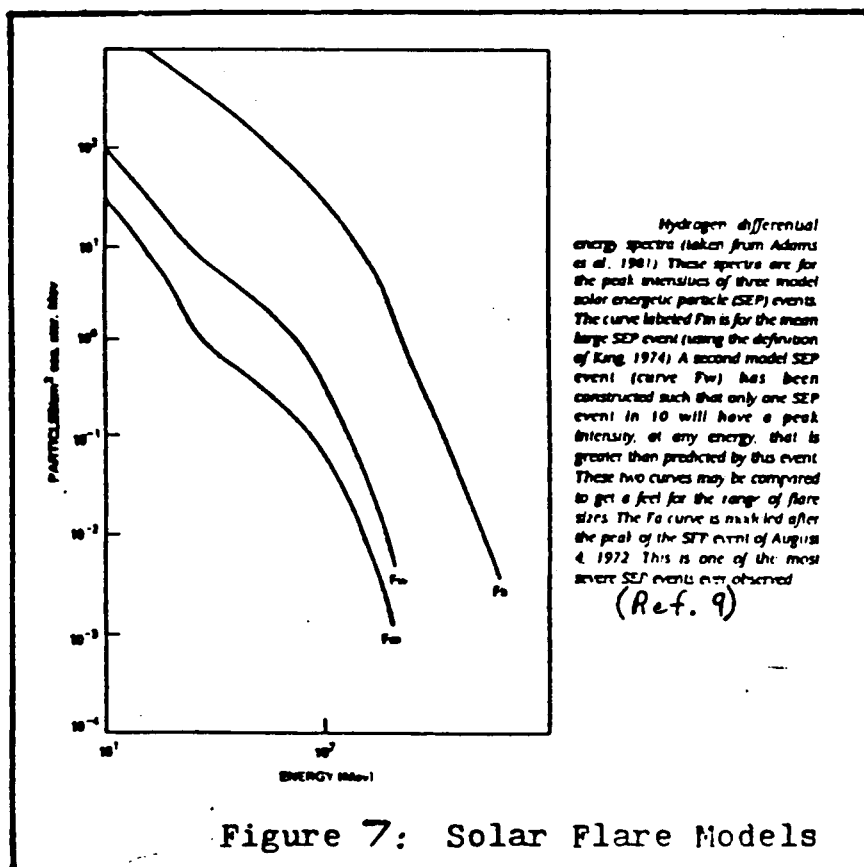
FIGURE 5: VEHICLE ASSEMBLY TENT (VAT)



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The attenuation of the annual dose due to cosmic-ray nuclei with shielding. The upper and lower curves show the dose equivalent and absorbed dose rates, respectively.

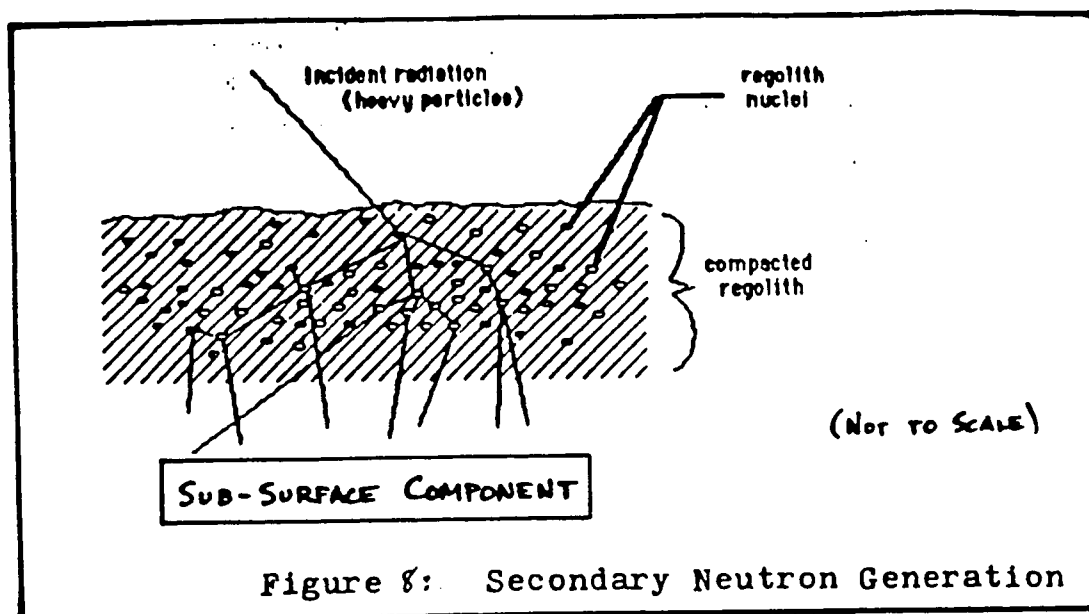
FIGURE 6: DOSE DUE TO COSMIC-RAY NUCLEI WITH SHIELDING (REF. 9)



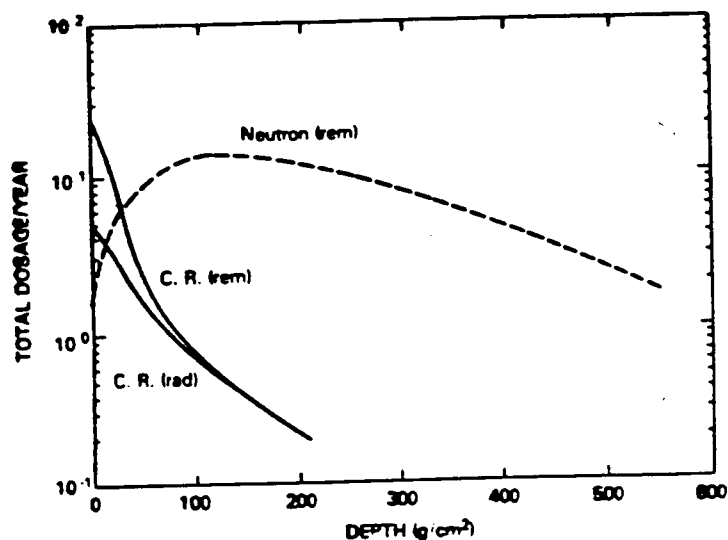
Hydrogen differential energy spectra (taken from Adams et al., 1981). These spectra are for the peak intensities of three model solar energetic particle (SEP) events. The curve labeled P1 is for the mean large SEP event (using the definition of King, 1974). A second model SEP event (curve P2) has been constructed such that only one SEP event in 10 will have a peak intensity, at any energy, that is greater than predicted by this event. These two curves may be compared to get a feel for the range of flare sizes. The P3 curve is marked after the peak of the SEP event of August 4, 1972. This is one of the most severe SEP events ever observed.

(Ref. 9)

Figure 7: Solar Flare Models

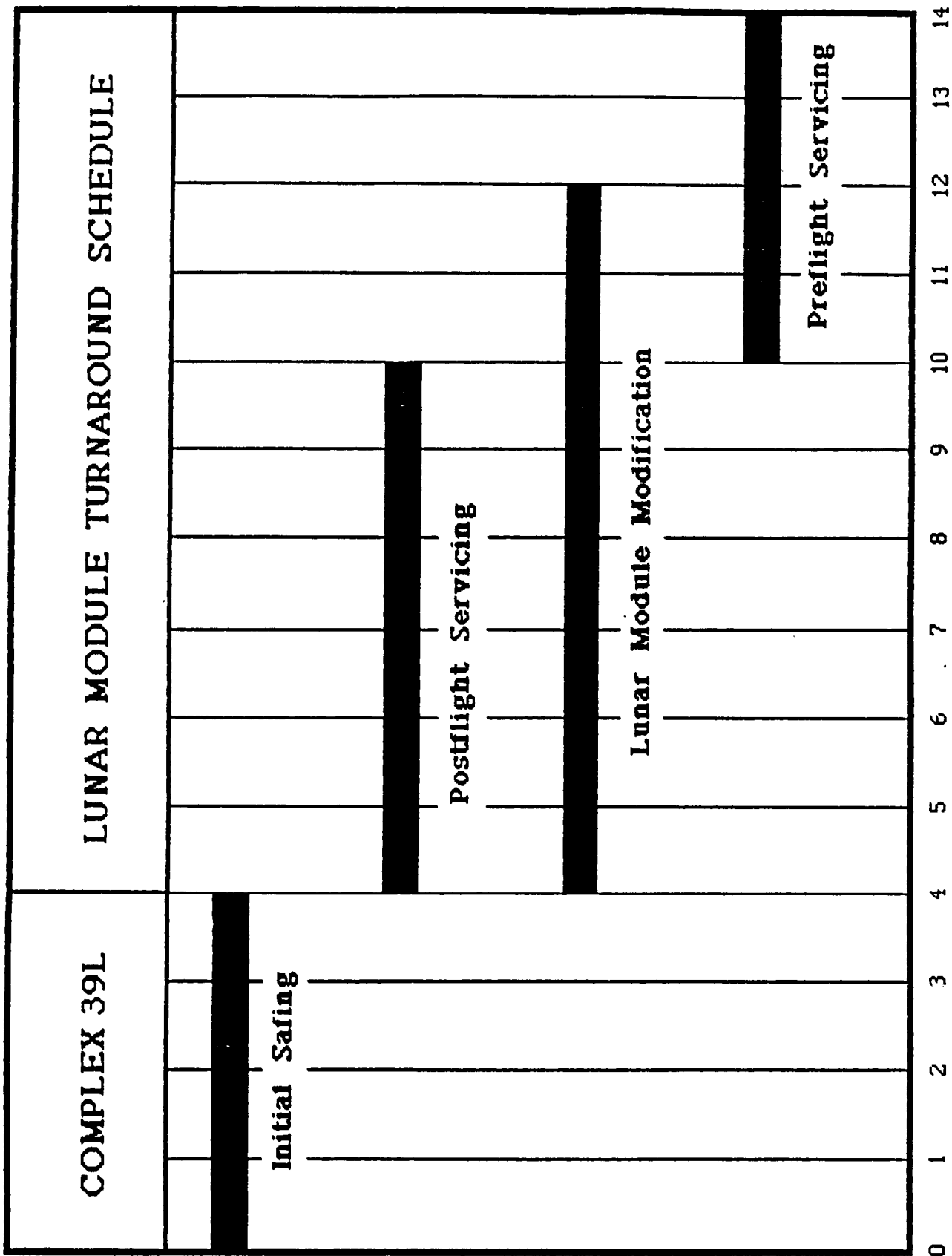


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A comparison of the annual dose equivalent due to secondary neutrons and cosmic-ray nuclei, as a function of shielding. Also, the absorbed dose rate due to cosmic-ray nuclei is shown.

FIGURE 9 : DOSE FROM SECONDARY NEUTRONS AND COSMIC-RAY NUCLEI (Ref. 9)



LANDING

TIME (DAYS)

LAUNCH

FIGURE 10: LUNAR MODULE TURNAROUND SCHEDULE

NOMENCLATURE

LLLF = Lunar Launch and Landing Facility

LEO = Low Earth Orbit

LLO = Low Lunar Orbit

FIT = Fuel Inventory Tent

VAT = Vehicle Assembly Tent

GCR = Galactic Cosmic-Rays

SEP = Solar Energy Particles

LPS = Launch Processing System

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EEI Report #87-172, December 1987.

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APPENDIX 6

Bridge Crane Design

ME 3073

Mechanical Systems Design I

Instructor: Dr. A.L. DiPare

Author: Jeff Dupper #79315

Date Submitted: 5/26/88

Introduction:

This report contains the design calculations and drawings for a bridge crane to be used by NASA on the moon for the proposed Lunar Landing and Launch Facility (LLLF or Complex 39L) in the year 2005 - 2010. This report is limited to a detailed analysis of the structure required for the crane. The actual lifting mechanism has been sketched to show how it will operate on the monorail, however, no design calculations were performed on that aspect of the design.

The main aspects of the analysis include selection of materials, design of an arch, design of the column supports, design of the monorail beam, cable selection, and an analysis of the stresses, strains, moments, and cross sectional properties of the entire structure. Also, a finite element analysis was performed to help analyze the structure given variable loading conditions.

Design Criteria:

The bridge crane is to be able to fit in a Quonset tent on the moon that forms a semicircle with a radius of 18 meters. The crane must be able to move the payload to any point in a volume defined by a width of 12 meters, length of 30 meters, and a height of 15 meters. The maximum weight of the payload is to be 44 metric earth tons.

The performance index for this crane is defined as (the weight of the payload) x (the span squared) divided by the weight of the structure itself. Therefore, since the weight

of the payload and the span are constants, the challenge is to make the structure of minimum weight. This is due to the tremendous cost of transporting materials from earth to the moon.

Design of Structure:

There are many methods of solving this problem, and some of the major ones were examined before the final analysis was completed. Due to the shape of the Quonset tent, a design was chosen that would exploit the naturally stronger geometry of an arch. The support beam acts as the monorail for the lifting mechanism as well as a tie rod for the arch to reduce bending moments at the supports. The support beam has cables symmetrically positioned to better support the weight, and allow for decreased weight of the beam.

Arch:

The use of arches for construction can be traced back many centuries, to before early Roman days when it was discovered that curved arches made of adjoining stones could span many times the distance of straight stone even if the stones in the arch were unbounded. "Since the strength of such arches does not seem to depend on the strength of the connecting joints, it must be that the arch is basically in compression throughout its entire arc length." (Zuk)

By a consideration of the force flow or pressure lines of an arch, it can be shown that the optimum shape of an arch

that is to carry a load is a parabola. In a parabola, the forces along the arch are in pure compression, smallest at the crown and increasing as they approach the supports.

In the bridge crane design, the arch is in the shape of an ellipse due to height problems. A parabola could not be used to obtain the span necessary without going outside of the Quonset tent.

The theory used to analyze the stresses in the arch were based on a circular member with the load at the top center. This method gives a very good approximation for the maximum stress since the arch of an ellipse can be very nearly approximated by the arc length of a circle with a relatively large radius. The load being in the center produces the greatest bending moment. The Winkler-Bach theory for curved beams was used to analyze the arch and can be found in the Standard Handbook for Mechanical Engineers (see reference).

Support Beam:

The support beam for this structure acts as a tie rod for the crane, a monorail for the lifting mechanism, as well as the support structure for lifting the payload. The design for the beam includes three cable supports symmetrically located along the length of the beam. Therefore, if the deflection in the cables is minimal, then the analysis of the beam is such that each section of the beam can be considered independently.

Columns:

The columns must be able to prevent tipping as well as support the stresses caused by the payload. Due to the use of an arch, the vertical load on the columns is increased and for the most part is in compression. In order to try to minimize the weight as well as provide the lateral support necessary, the columns were designed as two beams on each side of the structure located five meters apart. Also, a plate is used at the midsection to prevent bowing of the columns.

Materials Selection:

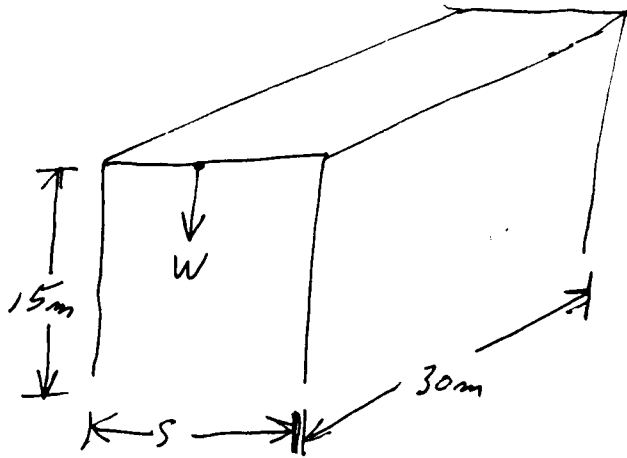
There were three major concerns in selecting the appropriate materials for this structure. The first and foremost was the weight consideration, second were the properties of the material at low temperatures due to the lunar atmosphere, and finally was high strength.

The main material used in the structure was a Titanium alloy. Although Titanium is relatively expensive, when weight is such an important factor, the cost of the material does not matter. The alloy used is Ti 6% Al-4% V, which is an alpha-beta Titanium alloy. The reason that Titanium is such a good choice for this design is that the mechanical properties improve when the temperature decreases. Titanium is a hexagonal-close-packed metal that exhibit mechanical properties intermediate between those of face-center-cubic, and body-center-cubic. Titanium and its alloys exhibit good

ductility at low temperatures, and are used extensively for applications for which weight reduction and low temperatures are necessary.

The beam is to be constructed of Mg 8.5% Al. This is because the maximum stress in the beam is relatively low, and the weight density of this Magnesium alloy is very low.

The weight of the cable is relatively little (3 Kg) and therefore, a standard grade of high strength steel wire is used.

DESIGN CRITERIA:

My Birthday
3/15

$$W = \overset{29}{\cancel{29}} \text{ metric tons on earth} + \# \text{ of B.D.}$$

$$\Rightarrow \text{mass} = \cancel{29}000 + 1500 = 44,000 \text{ Kg}$$

$$\Rightarrow \text{Weight} = \frac{1}{6} (44,000)(9.81) = \underline{\underline{71940 \text{ N}}}$$

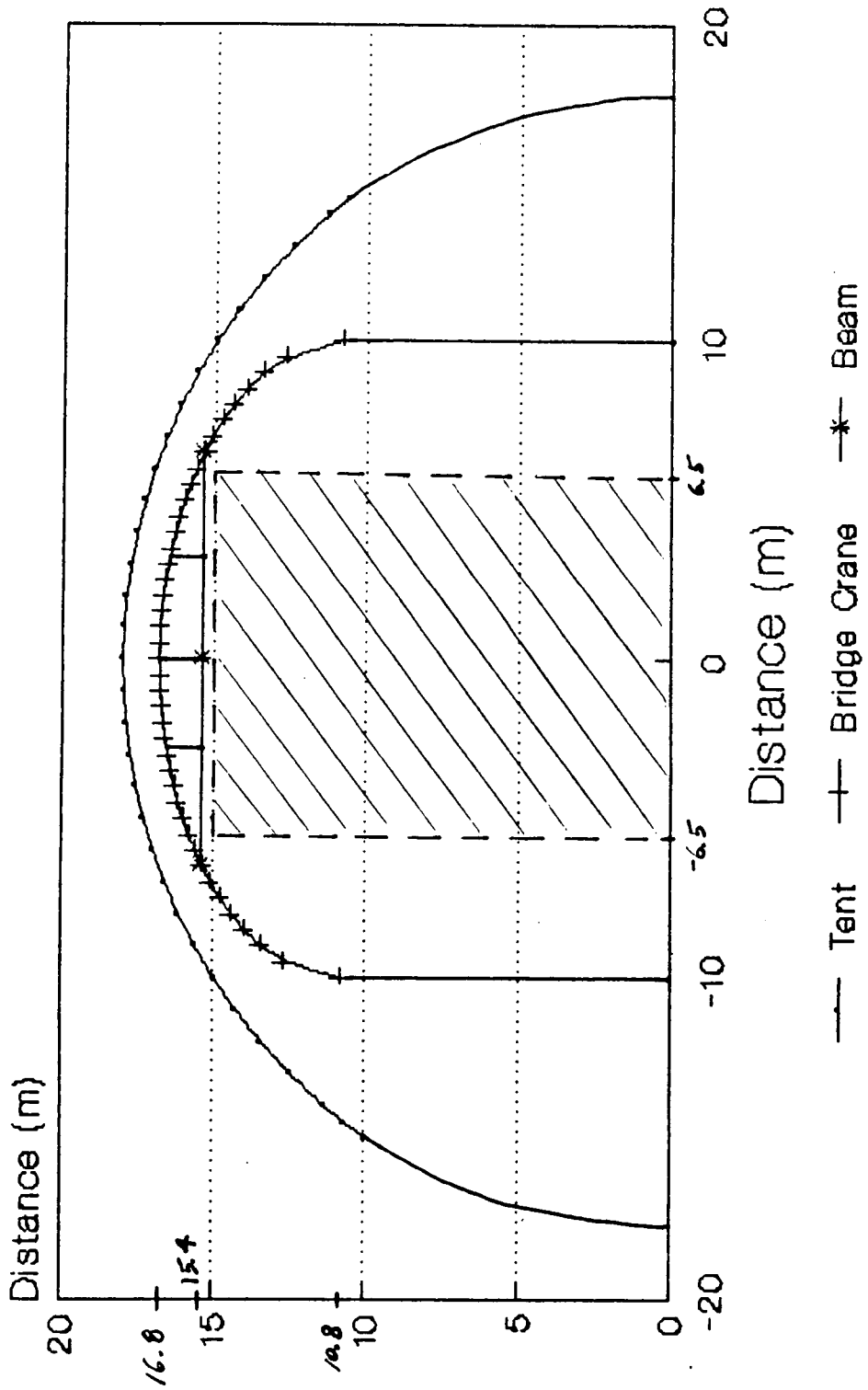
$$\text{Span} = 9 + \# \text{ Birth month} = 9 + 3$$

$$\Rightarrow \underline{\underline{\text{Span} = 12 \text{ m}}}$$

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Tent and Crane Mathematical Model



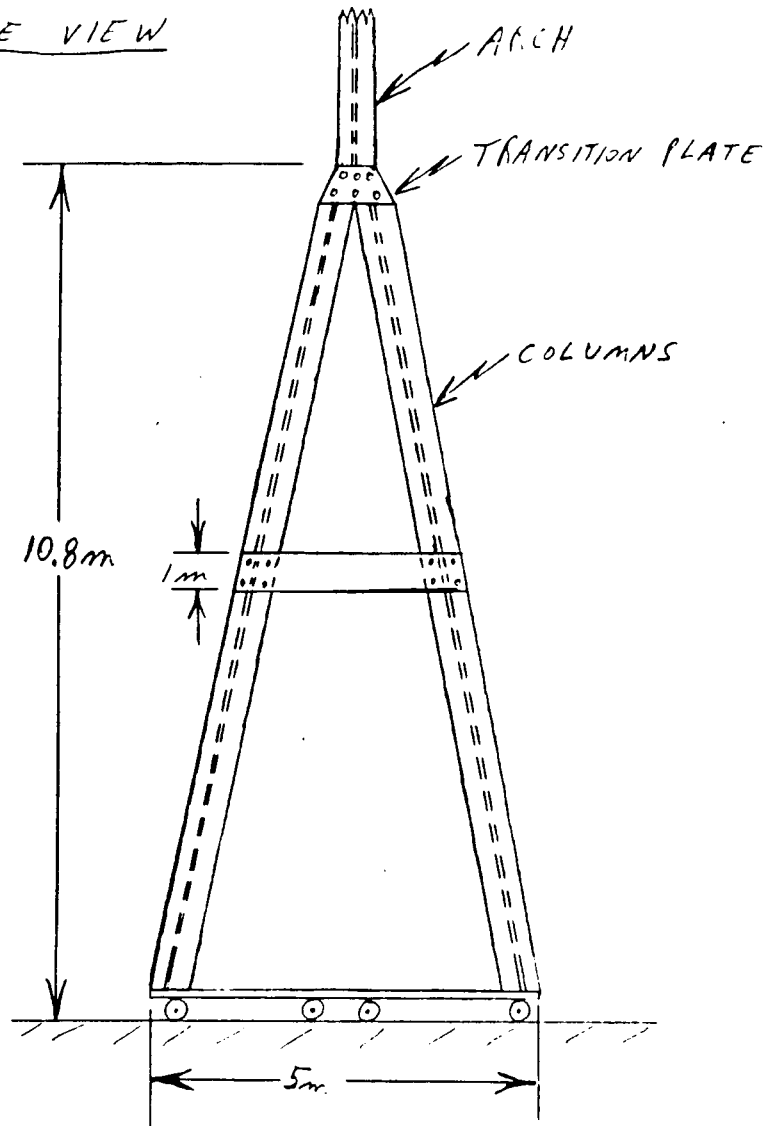
Jeff Dupper #79315

FRONT VIEW

SIDE VIEW

SCALE: 1cm = 1m

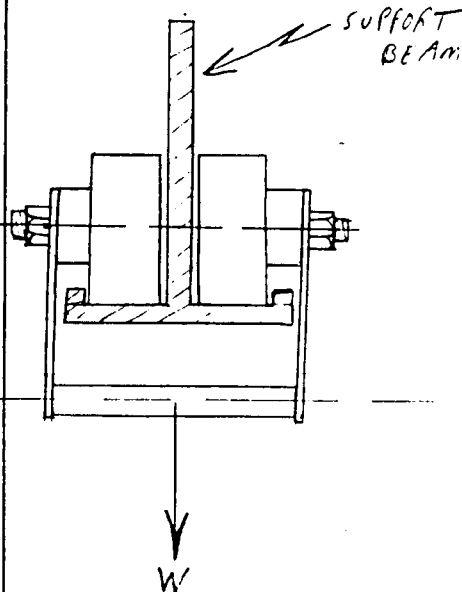
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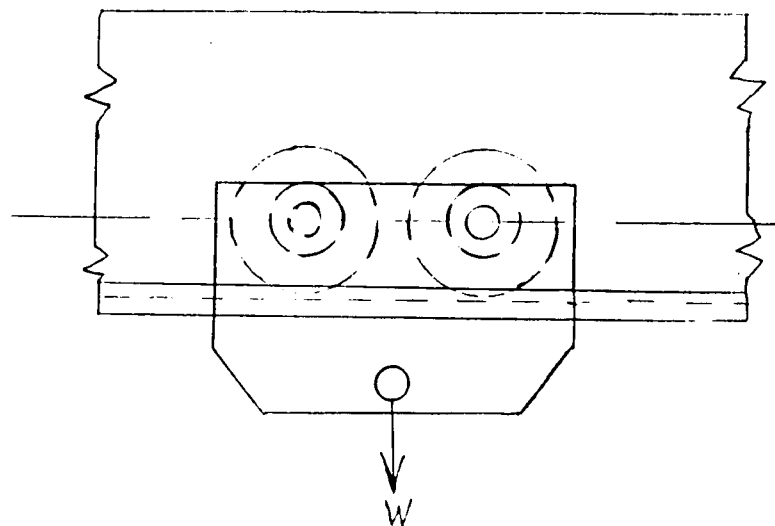
DETAIL OF MONORAIL:

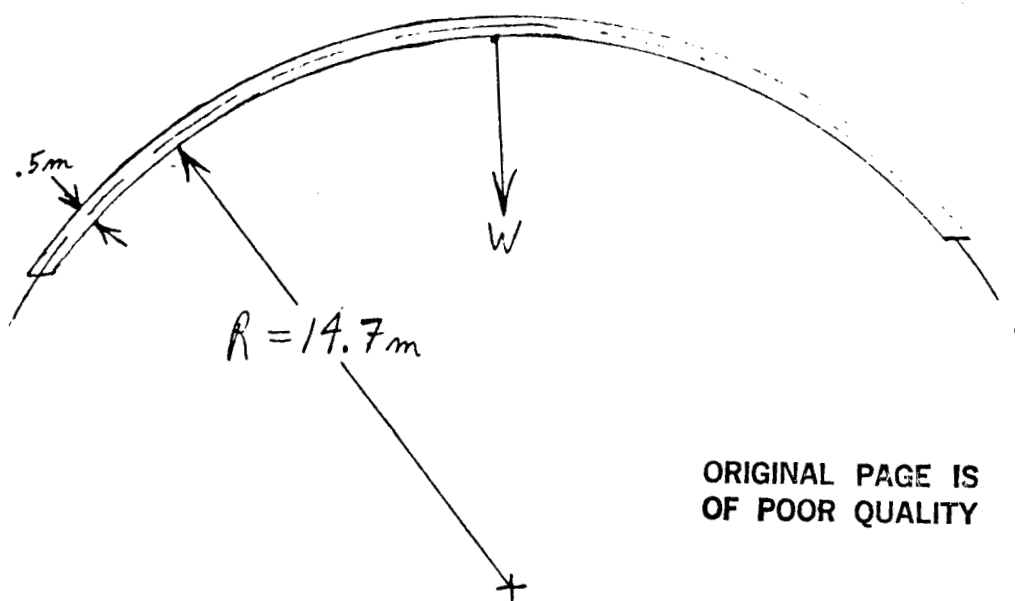
SCALE: 1cm = .1m

FRONT VIEW



SIDE VIEW



DESIGN OF ARCH :SCALE $\Rightarrow 1 \text{ cm} = 2 \text{ m}$ CROSS SECTION AND MATERIAL PROPERTIES:MATERIAL : ALPHA-BETA TITANIUM ALLOY (T_i - 6% AL - 4% V)

$$UTS = 150,000 \text{ psi} \Rightarrow 1017 \text{ MPa}$$

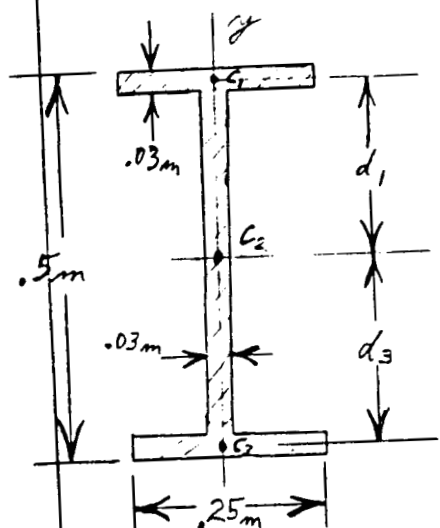
$$E = 16.5 \times 10^6 \text{ psi} \Rightarrow 114 \times 10^9 \text{ Pa}$$

$$YS = 140,000 \text{ psi} \Rightarrow 932 \text{ MPa}$$

$$f = 4460 \text{ kg/m}^3$$

$$\% \text{ ELONGATION} = 8$$

$$\text{SCALE} \Rightarrow 1 \text{ cm} = .1 \text{ m}$$



$$I_x = \sum I + Ad^2 = \sum \left(\frac{1}{12} b h^3 + A d^2 \right)$$

$$I = \frac{1}{12} b_1 h_1^3 + A_1 d_1^2 + \frac{1}{12} b_2 h_2^3 + A_2 d_2^2 + \frac{1}{12} b_3 h_3^3 + A_3 d_3^2$$

$$b_1 = b_3 = .25$$

$$b_2 = .03$$

$$h_1 = h_3 = .03$$

$$h_2 = .5 - 2(.03) = .44$$

$$A_1 = A_3 = (.25)(.03) = .0075 \text{ m}^2$$

$$A_2 = (.44)(.03) = .0132 \text{ m}^2$$

$$d_1 = d_3 = \frac{1}{2}(.5) - \frac{.03}{2} = .235 \text{ m} \quad d_2 = 0$$

$$\Rightarrow I_x = 2 \left(\frac{1}{12} (.25)(.03)^3 + (.0075)(.235)^2 \right) + \frac{1}{12} (.03)(.44)^3 + (.0132)(0)$$

$$\Rightarrow I_x = .0008295 + .000213$$

$$\boxed{I_x = .0010425 \text{ m}^4}$$

DESIGN OF COLUMNS:ORIGINAL PAGE IS
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MATERIAL: SAME AS ARCH

 $T_i: -6\%AL - 4\%V$ $Y_S = 932 \text{ MPa}$ $E = 114000 \text{ MPa}$ $f = 4460 \frac{\text{kg}}{\text{cm}^2}$

$$I_x = \sum I + Ad^2 \Rightarrow I_x = \sum \left(\frac{1}{12} b h^3 + A d^2 \right)$$

$$I_x = \frac{1}{12} b_1 h_1^3 + A_1 d_1^2 + \frac{1}{12} b_2 h_2^3 + A_2 d_2^2 + \frac{1}{12} b_3 h_3^3 + A_3 d_3^2$$

$$b_1 = b_3 = .5 \text{ m}$$

$$b_2 = .03 \text{ m}$$

$$h_1 = h_3 = .03 \text{ m}$$

$$h_2 = .44 \text{ m}$$

$$A_3 = (.44)(.03) = .0132 \text{ m}^2$$

$$A_1 = A_3 = (.03)(.5) = .015 \text{ m}^2 \quad d_3 = 0$$

$$d_1 = d_3 = .5 \left(\frac{1}{2} \right) - .03 \left(\frac{1}{2} \right) = .235 \text{ m}$$

$$\therefore I_x = 2 \left(\frac{1}{12} (.5)(.03)^3 + (.015)(.235)^2 \right) + \frac{1}{12} (.03)(.44)^3 + 0$$

$$\Rightarrow I_x = .001872 \text{ m}^4 \text{ FOR EACH COLUMN}$$

$$\therefore I_x = 2(.001872) = \underline{\underline{.003744 \text{ m}^4}} \text{ FOR BOTH COLUMNS (ONE SIDE)}$$

DESIGN OF SUPPORT BEAM: SCALE: 1cm = .1m

MEMBER	AREA	\bar{Y}	$\bar{Y} \cdot A$
1	.0111 m ²	.215 m	.00239
2	.009 m ²	.015 m	.000135

$$\Sigma A = .0201 \text{ m}^2 \quad \Sigma \bar{Y} A = .0025215 \text{ m}^3$$

$$\bar{Y} \Sigma A = \Sigma \bar{Y} A$$

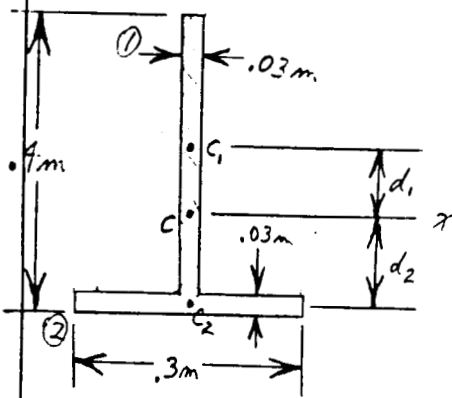
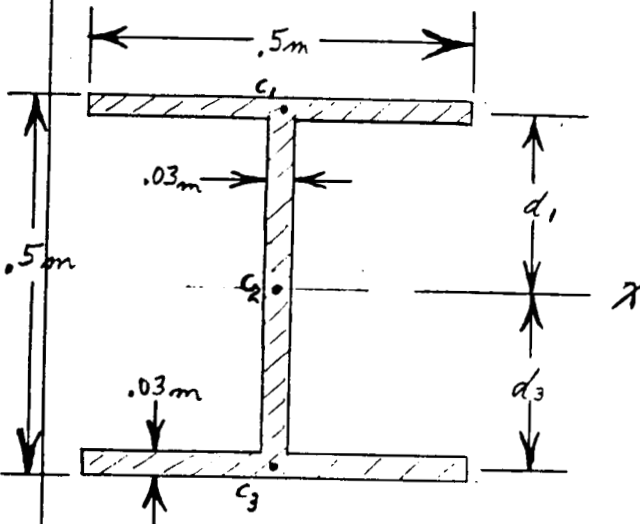
$$I_x = \sum (I + Ad^2)$$

$$\Rightarrow \bar{Y} = \frac{.0025215}{.0201} = .12545 \text{ m}$$

$$\Rightarrow I_x = \frac{1}{12} b_1 h_1^3 + A_1 d_1^2 + \frac{1}{12} b_2 h_2^3 + A_2 d_2^2$$

$$I_x = \frac{1}{12} (.03)(.37)^3 + (.0111)(.08955)^2 + \frac{1}{12} (.3)(.03)^3 + (.009)(.11045)^2$$

$$\Rightarrow \underline{\underline{I_x = .000326 \text{ m}^4}}$$



$$b_1 = .03 \text{ m}$$

$$b_2 = .3 \text{ m}$$

$$h_1 = .37 \text{ m}$$

$$h_2 = .03 \text{ m}$$

$$A_1 = .0111 \text{ m}^2$$

$$A_2 = .009 \text{ m}^2$$

$$d_1 = .215 - .12545 = .08955 \text{ m}$$

$$d_2 = .11045 \text{ m}$$

CABLE DESIGN:

MATERIAL: IMPROVED PLow WIR.

$$\sigma_{all} = \frac{1379}{2} = \underline{\underline{690 \text{ MPa}}}$$

$$E = 100000 \text{ MPa}$$

$$\text{BREAKING STRESS} = 1379 \text{ MPa}$$

$$F.S. = 2$$

$$\rho = 7860 \frac{\text{kg}}{\text{m}^3}$$

$$\Rightarrow d = \left(\frac{4F}{\sigma \pi} \right)^{1/2}$$

* DESIGN TO SUPPORT TOTAL
PAYLOAD WEIGHT = 71940 N = F

$$d = \left(\frac{4(71940)}{(690 \times 10^6) \pi} \right)^{1/2} = \underline{\underline{.0115 \text{ m}}}$$

$$I_x = \frac{1}{4} \pi r^4 = \frac{1}{4} \pi \left(\frac{.0115}{2} \right)^4$$

$$I_x = \underline{\underline{8.59 \times 10^{-10} \text{ m}^4}}$$

NEED 3 LENGTHS OF WIRE (SEE FIG "BEAM ANALYSIS")

$$L_1 = L_3 = 16.4205 - 15.3596 = \underline{\underline{1.0609 \text{ m}}}$$

$$L_2 = 16.8 - 15.3596 = \underline{\underline{1.4404 \text{ m}}}$$

DEFLECTION IN CABLES:

$$\delta = \frac{Pl}{AE}$$

$$P = 71940 \text{ N}$$

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$$L_1 = L_3 = 1.0609 \text{ m}$$

$$L_2 = 1.4404 \text{ m}$$

$$\delta_1 = \delta_3 = \frac{(71940)(1.0609)}{(1.037 \times 10^{-8})(100000)}$$

$$A = 1.037 \times 10^{-8} \text{ m}^2$$

$$E = 100000 \text{ N/m}^2$$

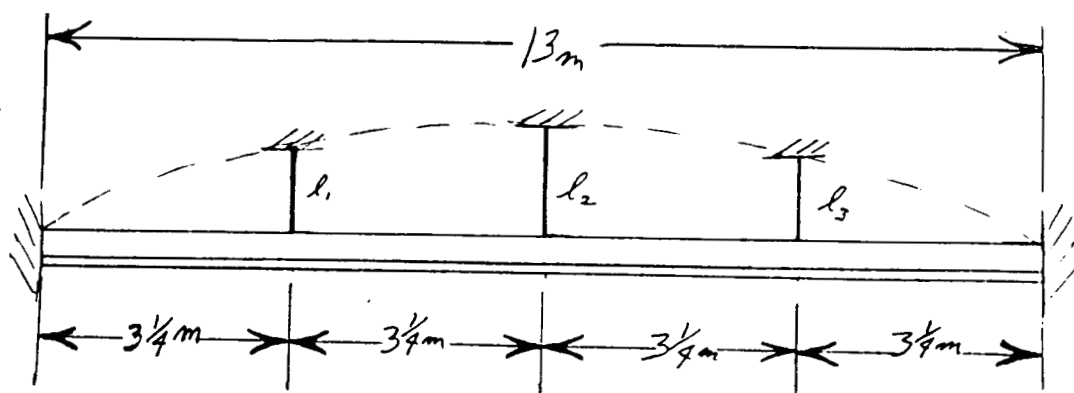
$$\Rightarrow \underline{\underline{\delta_1 = \delta_3 = .00736 \text{ m}}}$$

$$\delta_2 = \frac{(71940)(1.4404)}{(1.037 \times 10^{-8})(100000)}$$

* THESE REPRESENT
MAXIMUM DEFLECTIONS

$$\Rightarrow \underline{\underline{\delta_2 = .00999 \text{ m}}}$$

BEAM ANALYSIS: Scale; 1cm = 1m



$$I = .000326 \text{ m}^4$$

$$C = .12545 \text{ m}$$

$$\sigma_{\max} = \frac{M_{\max} C}{I}$$

* DUE TO THE SUPPORTING CABLES, EACH LENGTH OF THE BEAM WILL BE CONSIDERED A SIMPLY SUPPORTED BEAM. \therefore MAX BENDING MOMENT WILL OCCUR AT 1.625m, 4.875m, 8.125m, & 11.375m AND WILL ALL BE EQUAL.

$$W = 71940 \text{ N}$$

$$M_{\max} = (1.625)(71940) = 116903 \text{ N}\cdot\text{m}$$

$$\therefore \sigma_{\max} = \frac{116903(.12545)}{.000326} = 45 \text{ MPa}$$

MATERIAL: MAGNESIUM ALLOY (8.5% AL)

$$\rho = 1800 \frac{\text{kg}}{\text{m}^3} \quad \text{U.T.S.} = 380 \text{ MPa}$$

$$\text{Y.S.} = 275 \text{ MPa}$$

$$\therefore \text{F.S.} = \frac{275}{45} = \underline{\underline{6}}$$

ANALYSIS OF ARCH:

Reference:

ASSUME WORSE CASE:
LOAD IS ACTING AT
CENTER OF ARCH. THIS
PRODUCES THE MAXIMUM MOMENT.

Standard Handbook for
Mechanical Engineers
Baumeister & Marks 7th Edition

ALSO, ASSUME FULL LOAD OF PAYLOAD.

THEORY:DATA:

$$R_i = 14.45 \text{ m}$$

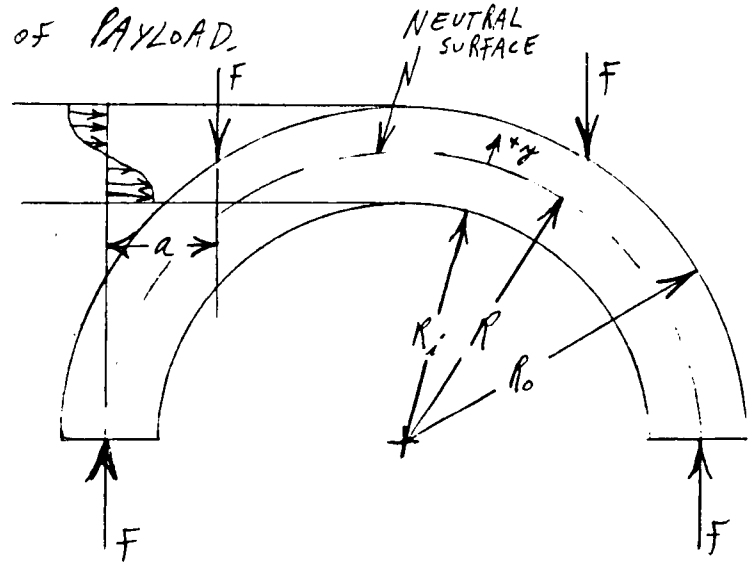
$$R = 14.7 \text{ m}$$

$$R_o = 14.95 \text{ m}$$

$$y = .25 \text{ m}$$

$$a = 10 \text{ m}$$

$$F = W = 71940 \text{ N}$$



FORMULAS: FOR THE STRESS AT A
POINT y UNITS FROM THE
CENTROIDAL AXIS (C.A.):

$$S = \frac{M}{AR} \left[1 + \frac{y}{z(R+y)} \right]$$

$M \equiv$ BENDING MOMENT

$y \equiv$ DISTANCE FROM C.A.

$A \equiv$ CROSS SECTIONAL AREA

$R \equiv$ RADIUS OF CENTROIDAL AXIS

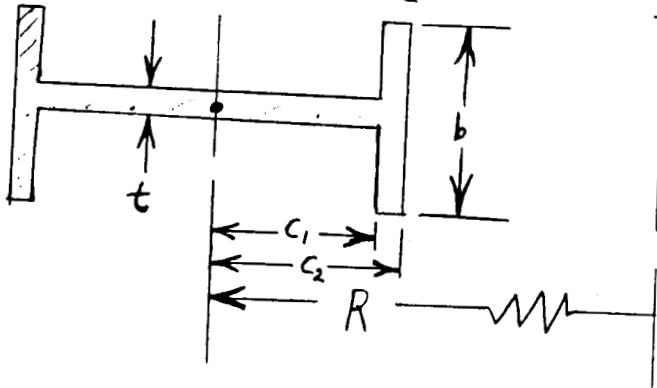
$z \equiv$ CROSS SECTIONAL PROPERTY $\Rightarrow z = -\frac{1}{A} \int \frac{y}{R+y} dA$

ANALYSIS OF ARCH CONTINUED:

FOR THIS CROSS SECTION,

$$Z = -\frac{1}{A} \int \frac{y}{R+y} dA = -1 + \frac{R}{A} \left[b \ln \left(\frac{R+C_2}{R-C_2} \right) + (t-b) \ln \left(\frac{R+C_1}{R-C_1} \right) \right]$$

$$\text{WHERE } A = 2[(t-b)C_1 + bC_2]$$



$$R = 14.7 \text{ m}$$

$$C_1 = .22 \text{ m}$$

$$C_2 = .25 \text{ m}$$

$$t = .03 \text{ m}$$

$$b = .25 \text{ m}$$

CALCULATE A:

$$A = 2[(.03-.25)(.22) + (.25)(.25)] = \underline{\underline{.0282 \text{ m}^2}}$$

CALCULATE Z:

$$Z = -1 + \frac{14.7}{.0282} \left[.25 \ln \left(\frac{14.7+.25}{14.7-.25} \right) + (.03-.25) \ln \left(\frac{14.7+.22}{14.7-.22} \right) \right]$$

$$\Rightarrow Z = -1 + 521.3 [.008504 - .006586]$$

$$\therefore \underline{\underline{Z = .0001006}}$$

CALCULATE MAX STRESS:

$$S = \frac{M}{AR} \left[1 + \frac{Z}{R+Z} \right]$$

$$\Rightarrow S = \frac{719400}{(.0282)(14.7)} \left[1 + \frac{.0001006}{(.0001006)(14.7+.25)} \right]$$

$$\Rightarrow S = 1735418 [167.23]$$

$$\therefore S = 290208260 \text{ Pa} = \underline{\underline{290 \text{ MPa}}}$$

$$M = F \cdot d = 71940(10)$$

$$M = 719400 \text{ N} \cdot \text{m}$$

$$y = .25 \text{ m}$$

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ANALYSIS OF AFCH CONTINUED:

CALCULATE FACTOR OF SAFETY:

$$S_y = 932 \text{ MPa}$$

$$s_{max} = 290 \text{ MPa}$$

$$F.S. = \frac{S_g}{S_{max}} = \frac{932}{290}$$

$$\Rightarrow \underline{\underline{F.S. = 3.21}}$$

ANALYSIS OF FRAME STRUCTURE USING BEAM AS A TIE ROD.

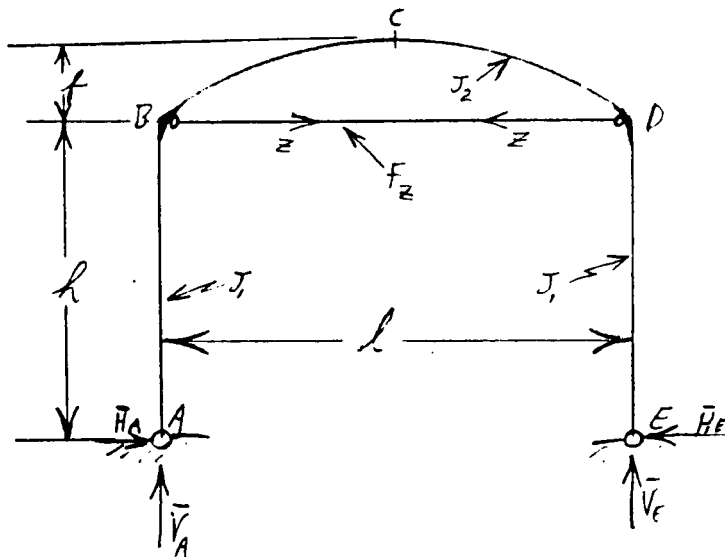
ASSUMPTIONS: FRAME IS 2 HINGED

SYMMETRICAL WITH PARABOLIC GIRDER.

*ALTHOUGH THIS ANALYSIS IS FOR A PARABOLIC STRUCTURE, IT IS VERY NEARLY EQUAL TO THE ELLIPSOIDAL ARCH USED.

REFERENCE: RIGID

FRAME ANALYSIS
FORMULAS, KLEINLOGEL



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CALCULATE

$$Z \equiv \text{TENSION IN THE ROD}$$
$$\bar{M}_B \equiv \text{MOMENT AT JOINT E}$$
$$\overline{M}_c \equiv \text{Moment At Joint C}$$
$$\bar{M}_D \equiv \text{MOMENT AT JOINT D}$$
$$\bar{H}_A \equiv \text{HORIZONTAL REACTION AT A}$$
$$\hat{H}_E \equiv \text{HORIZONTAL REACTION AT E}$$
$$\bar{V}_A = \text{VERTICAL REACTION AT A}$$
$$\bar{V}_E \equiv \text{VERTICAL REACTION AT E}$$

FRAME ANALYSIS CONTINUED:

STEP 1) PERFORM CALCULATIONS AS IF NO TIE ROD WAS PRESENT.

COEFFICIENTS:
(See Fig. Previous Page)

$$K = \frac{J_2}{J_1} \cdot \frac{L}{l}$$

$J_1 \equiv$ MOMENT OF INERTIA OF VERTICAL COLUMN

$J_2 \equiv$ MOMENT OF INERTIA OF ARCH

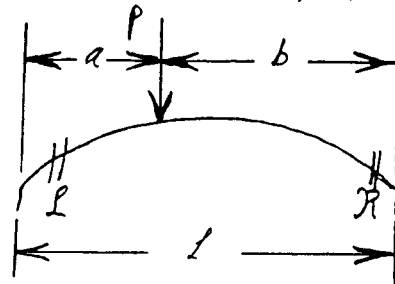
$$\Phi = \frac{f}{R}$$

$$B = 2K + 3 + 2\Phi$$

$$C = 2\Phi \left(1 + \frac{4}{5}\Phi\right)$$

$$N = B + C$$

* CALCULATION FOR SINGLE SYMMETRICAL LOAD AT CENTER TOP OF ARCH. LOAD IS EQUAL TO PAYLOAD.



L & R ARE LOAD FACTORS:

$$L = P a \beta (1 + \beta) \quad \alpha = \frac{a}{L}$$

$$R = P b \alpha (1 + \alpha) \quad \beta = \frac{b}{L}$$

FOR CONCENTRATED CENTER LOADS:

$$L = R = P \left(\frac{L}{2}\right) \left(\frac{1}{2}\right) \left(1 + \frac{1}{2}\right)$$

$$\Rightarrow \underline{\underline{L = R = PL \left(\frac{3}{8}\right)}}$$

$$\underline{\underline{G_1 = G_2 = \frac{1}{2} PL}}$$

$$M_B = M_D = - \frac{(L + R) + \Phi D}{2N}$$

$$M_C = M_C^0 + (1 + \Phi) M_B$$

$$H_A = H_E = - \frac{M_E}{R}$$

$$V_A = \frac{G_1}{L}$$

$$V_E = \frac{G_2}{L}$$

$$M_C^0 = \frac{Pl}{4}$$

$$D = \frac{24}{L^3} \int_0^L M_x^0 \cdot x x' dx$$

$$\Rightarrow (L + R) - \frac{24}{L^3} T$$

$$\Rightarrow \underline{\underline{D = \frac{5}{8} Pl}} \quad \text{FOR LOAD CASE DESCRIBED}$$

FRAME ANALYSIS CONTINUED:SUBSTITUTION FOR STEP 1:

$$P = 71940 \text{ N} = W$$

$$J_1 = .003744 \text{ m}^4 \text{ (see "DESIGN OF COLUMNS")}$$

$$J_2 = .0010425 \text{ m}^4 \text{ (see "DESIGN OF ARCH")}$$

$$R = 10.8 \text{ m}$$

$$l = 20 \text{ m}$$

$$f = 6 \text{ m}$$

$$K = \frac{J_2}{J_1} \cdot \frac{R}{l} = \left(\frac{.0010425}{.003744} \right) \left(\frac{10.8}{20} \right) = \underline{\underline{.1504}}$$

$$Q = \frac{f}{R} = \frac{6 \text{ m}}{10.8 \text{ m}} = \underline{\underline{.5555}}$$

$$B = 2K + 3 + 2Q = 2(.1504) + 3 + 2(.5555) = \underline{\underline{4.4119}}$$

$$C = 2Q(1 + \frac{4}{3}Q) = 2(.5555)(1 + \frac{4}{3}(.5555)) = \underline{\underline{1.6047}}$$

$$N = B + C = 4.4119 + 1.6047 = \underline{\underline{6.0166}}$$

$$L = R = Pl(\frac{3}{8}) = (71940)(20)(\frac{3}{8}) = \underline{\underline{539550 \text{ MPa} \cdot \text{m}}}$$

$$G = \frac{1}{2} Pl = \frac{1}{2}(71940)(20) = \underline{\underline{719400 \text{ MPa} \cdot \text{m}}}$$

$$D = \frac{5}{8} Pl = \frac{5}{8}(71940)(20) = \underline{\underline{899250 \text{ MPa} \cdot \text{m}}}$$

$$M_C = \frac{Pl}{4} = \frac{(71940)(20)}{4} = \underline{\underline{359700 \frac{\text{MPa}}{\text{m}}}}$$

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FRAME ANALYSIS CONTINUED:DETERMINE MOMENTS & REACTIONS ASSUMING NO TIE ROD:

$$M_B = M_D = - \frac{(L+K) + QP}{2N} = - \frac{2(539550) + (.5555)(899250)}{2(6.0166)}$$

$$\Rightarrow \underline{M_B = M_D = -131190 \text{ MPa} \cdot \text{m}}$$

$$M_C = M_C^0 + (1+Q)M_B = 359700 + (1+.5555)(-131190)$$

$$\Rightarrow \underline{M_C = 155634 \text{ MPa} \cdot \text{m}}$$

$$\underline{H_A = H_E = -\frac{M_B}{L} = \frac{131190}{10.8} = 12147 \text{ MN}}$$

$$\underline{V_A = V_E = \frac{G}{L} = \frac{719400}{20} = 35970 \text{ MN}}$$

STEP 2)CALCULATE ADDITIONAL COEFFICIENTS:

$$N_z = \frac{2(4K+1)}{N} + L \quad \beta = \frac{C}{N} \quad \gamma = \frac{QB-C}{N} \quad L = \frac{15\sqrt{2}}{2f_z^2} \cdot \frac{E}{E_z}$$

 $E = \text{MODULUS OF ELASTICITY FOR FRAME MATERIAL}$ $E_z = \text{MODULUS OF ELASTICITY FOR TIE ROD}$ $F_z = \text{CROSS-SECTIONAL AREA OF TIE ROD}$

$$E = 117000 \text{ MPa (SEE "DESIGN OF ARCH")}$$

$$E_z = 117000 \text{ MPa (SAME MATERIAL - SUPPORT FRAME)}$$

$$F_z = .020 / \text{m}^2$$

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FRAME ANALYSIS CONTINUED:STEP 2 CONTINUED:

$$\beta = \frac{C}{N} = \frac{1.6097}{6.0166} = .2667$$

$$\gamma = \frac{QB - C}{N} = \frac{(5555)(7.4119) - 1.6097}{6.0166} = .14063$$

$$L = \frac{15J_2 \cdot E}{2F^2F_z E_z} = \frac{15(.0010925)}{2(6)^2(.0201)} \cdot 1 = .01081$$

$$N_z = \frac{2(4K+1)}{N} + L = \frac{2(4(.1509)+1)}{6.0116} + .01081 = .5436$$

STEP 3) CALCULATE TENSION IN TIE ROD:

$$Z = \frac{\frac{M_B + M_D}{2} + 4(M_C - M_C^*) + \frac{5}{4} D}{f N_z}$$

$$\Rightarrow Z = \frac{-131190 + 4(155634 - 359700) + \frac{5}{4}(899250)}{(6)(.5436)}$$

$$\Rightarrow \underline{Z = 54148 \text{ MPa}}$$

STEP 4) CALCULATION OF MOMENTS & REACTIONS WITH TIE ROD:
(See Fig. "ANALYSIS OF FRAME")

$$\bar{M}_B = \bar{M}_D = M_B + \beta Z L = -131190 + .2667(54148)(10.8)$$

$$\Rightarrow \boxed{\bar{M}_B = \bar{M}_D = 24775 \text{ MPa} \cdot \text{m}}$$

$$\bar{M}_C = M_C - \gamma Z L = 155634 - .14063(54148)(10.8)$$

$$\Rightarrow \boxed{\bar{M}_C = 73394 \text{ MPa} \cdot \text{m}}$$

$$\bar{H}_A = \bar{H}_E = H_A - \beta Z = 12147 - .2667(54148)$$

$$\Rightarrow \boxed{\bar{H}_A = \bar{H}_E = -2294 \text{ kN}}$$

$$\boxed{\bar{V}_A = \bar{V}_E = 35970 \text{ MN}}$$

STRESSES ON FRAMING:

$$\sigma = \frac{MC}{I}$$

$$C_A = C_E = .235 \text{ m}$$

$$C_D = C_C = C_F = .235 \text{ m}$$

$$\sigma = \frac{F}{A} \text{ (COLUMNS)}$$

VERTICAL FORCES

$$I_A = I_E = .003744 \text{ m}^4$$

$$\tau = \frac{F}{A} \text{ (COLUMNS)}$$

HORIZONTAL FORCES

$$I_C = I_D = I_F = .0010425 \text{ m}^4$$

$$A_A = A_E = .0432 \text{ m}^2 \text{ EACH}$$

$$\sigma_A = \sigma_E = \frac{35970}{.0869} = 416319 \text{ MPa}$$

$$\tau_A = \tau_E = \frac{2294}{.0869} = 26551 \text{ MPa}$$

$$\sigma_B = \sigma_D = \frac{(24776)(.235)}{.0010425} = \underline{\underline{5.58 \text{ MPa}}}$$

$$\sigma_C = \frac{73394(.235)}{.0010425} = \underline{\underline{16.5 \text{ MPa}}}$$

∴ STRESSES ARE ACCEPTABLE IN FRAMING

SINCE $\sigma_{max} < \sigma_{allowable}$

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OF HIGH QUALITY

TOTAL MASS OF STRUCTURE :ARCH : ELLIPSOIDAL, LENGTH $\cong 21\text{ m}$

AREA OF CROSS SECTION = $.0282\text{ m}^2$

MASS DENSITY OF MATERIAL = 4460 kg/m^3

VOLUME = $.5922\text{ m}^3$

$\therefore \underline{\underline{\text{MASS} = 2641\text{ Kg}}}$

COLUMNS : LENGTH = 10.8 m each

AREA OF CROSS SECTION = $.0432\text{ m}^2$ each

MASS DENSITY OF MATERIAL = 4460 kg/m^3

VOLUME = $.93312\text{ m}^3$ FOR BOTH SIDES

$\therefore \underline{\underline{\text{MASS} = 4162\text{ Kg}}}$

SUPPORT BEAM : LENGTH = 13 m

AREA OF CROSS SECTION = $.0201\text{ m}^2$

MASS DENSITY OF MATERIAL = 1800 kg/m^3

VOLUME = $.2613\text{ m}^3$

$\therefore \underline{\underline{\text{MASS} = 470\text{ Kg}}}$

CABLES : TOTAL LENGTH = 3.5622 m

AREA OF CROSS SECTION = $1.037 \times 10^{-6}\text{ m}^2$

MASS DENSITY OF MATERIAL = 7860 kg/m^3

VOLUME = $3.77 \times 10^{-6}\text{ m}^3$

$\therefore \underline{\underline{\text{MASS} = 3\text{ Kg}}}$

TOTAL MASS :ORIGINAL PAGE IS
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$$M_{\text{TOT}} = 2641 + 4162 + 470 + 3 = \underline{\underline{7276\text{ Kg}}} \Rightarrow \underline{\underline{7137}}$$

CALCULATE PERFORMANCE INDEX:

$$P.I. = \frac{WS^2}{\text{Weight}} = \frac{(71940)(13)^2}{71377}$$

$$\Rightarrow \boxed{P.I. = 170.33}$$

References

- 1) Askeland Donald R., The Science and Engineering of Materials
- 2) Baumeister & Marks, Standard Handbook for Mechanical Engineers Seventh Edition
- 3) Beer and Johnston, Mechanics of Materials
- 4) Kleinlogel, Rigid Frame Formulas
- 5) Morris and Carpenter, Structural Frames
- 6) Parcel and Maney, Statically Indeterminate Stresses
- 7) Spotts, Design of Machine Elements 6 th Edition
- 8) Zuk William, Concepts of Structure

Finite Element Analysis

Software used: IMAGES 2D by Celestial Software Inc.

Copywrite 1983

Results of Finite Element Analysis

Three material properties were defined, one for the arch and columns, one for the support beam, and one for the cables. Also, the cross sectional properties were defined for each.

When reviewing the data from the finite element analysis, refer to the figures that precedes the data in the report. There, the locations of the nodes and elements can be seen. Restraints for the structure were given as fully restrained at node 66 and node 70. This is an accurate model since the support columns will be in a "locked" position when the payload is being raised or lowered. Also, this is a more conservative analysis. The loads given to this analysis were a single load at the center of the support beam (node 11) which was equal in magnitude to the weight of the payload, as well as a vertical load representing an acceleration caused by the movement of the lifting mechanism. This load was set at 5% of the total payload weight. These loads represent the worse case as far as stresses are concerned. With this model now complete, it is very easy to change the variables to examine different designs.

The maximum displacements, loads, and stresses have been highlighted in the report for easy reference. As can be seen by the results, the cables should be made with a larger diameter to decrease the amount of stretching (1.36cm). Based on the maximum stress at the support of 447.8 MPa, the overall factor of safety for the structure is 2.081.

CALCULATION OF POINTS FOR ELLIPSE:

GENERAL EQUATION: $\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1$

let $a = 10m$

let $b = 6m$

$$\Rightarrow \frac{x^2 b^2 + y^2 a^2}{a^2 b^2} = 1$$

$$\Rightarrow \frac{x^2}{100} + \frac{y^2}{36} = 1$$

$$\Rightarrow x^2 b^2 + y^2 a^2 = a^2 b^2$$

$$\Rightarrow y = \sqrt{b^2 - \frac{x^2 b^2}{a^2}}$$

* ONLY PLOT COORDINATES
IN FIRST & SECOND
QUADRANTS. $\therefore y$ IS POSITIVE ONLY

$$\Rightarrow y = (36 - \frac{x^2(36)}{100})^{\frac{1}{2}}$$

$$\Rightarrow y = (36 - .36x^2)^{\frac{1}{2}}$$

CALCULATION OF POINTS FOR CIRCULAR TENT: (NOT USED)

GENERAL EQUATION:

$$x^2 + y^2 = r^2$$

$$\Rightarrow y = \sqrt{r^2 - x^2}$$

let $r = 18m$

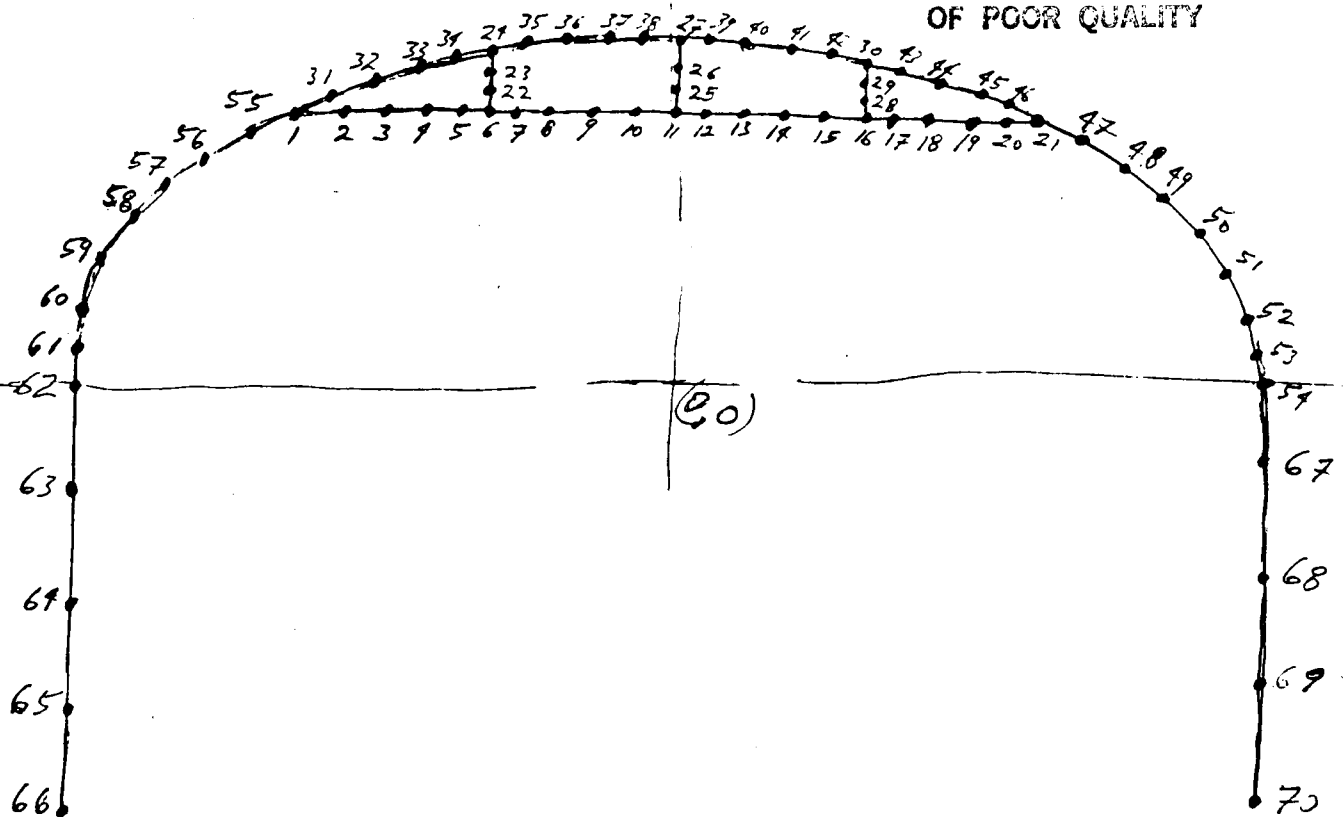
$$\Rightarrow y = \sqrt{324 - x^2}$$

POINTS FOR BEAM:

height = 15.4 m

lsp endpoint = 6.5m

rsp endpoint = 6.5m

NODES :ORIGINAL PAGE IS
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1 (-6.5, 4.56)	21 (6.5, 4.56)	41 (1.95, 5.8848)	61 (-9.4595, 1.9459)
2 (-5.85, 4.56)	22 (-3.25, 4.9314)	42 (2.6, 5.7937)	62 (-10, 0)
3 (-5.24, 4.56)	23 (-3.25, 5.3029)	43 (3.9, 5.5249)	63 (-10, 2.7)
4 (-4.55, 4.56)	24 (-3.25, 5.6743)	44 (4.55, 5.3429)	64 (-10, -5.4)
5 (-3.9, 4.56)	25 (0, 5.04)	45 (5.2, 5.1250)	65 (-10, -8.1)
6 (-3.25, 4.56)	26 (0, 5.52)	46 (5.85, 4.8662)	66 (-10, -10.8)
7 (-2.6, 4.56)	27 (0, 6.0)	47 (6.7568, 4.4232)	67 (10, -2.7)
8 (-1.95, 4.56)	28 (3.25, 4.9314)	48 (7.2973, 4.1024)	68 (10, -5.4)
9 (-1.3, 4.56)	29 (3.25, 5.3029)	49 (7.8378, 3.7262)	69 (10, -8.1)
10 (-.65, 4.56)	30 (3.25, 5.6743)	50 (8.3789, 3.2755)	70 (10, -10.8)
11 (0, 4.56)	31 (-5.85, 4.8662)	51 (8.6486, 3.012)	
12 (.65, 4.56)	32 (-5.2, 5.125)	52 (9.1892, 2.3667)	
13 (1.3, 4.56)	33 (-4.55, 5.3429)	53 (9.4595, 1.9459)	
14 (1.95, 4.56)	34 (-3.9, 5.525)	54 (10, 0)	
15 (2.6, 4.56)	35 (-2.6, 5.7937)	55 (-6.7568, 4.4232)	
16 (3.25, 4.56)	36 (-1.95, 5.8848)	56 (-7.2973, 4.1024)	
17 (3.9, 4.56)	37 (-1.3, 5.9491)	57 (-7.8378, 3.7262)	
18 (4.55, 4.56)	38 (-.65, 5.9873)	58 (-8.3789, 3.2755)	
19 (5.2, 4.56)	39 (.65, 5.9873)	59 (-8.6486, 3.012)	
20 (5.85, 4.56)	40 (1.3, 5.9491)	60 (-9.1892, 2.3667)	

ELEMENTS:ORIGINAL PAGE IS
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(NODE 9, NODE 6) GROSS SECTION, MATERIAL

1) 1,2,3,2	22) 65,64,2,1	43) 27,39,1,1	64) 69,70,2,1
2) 2,3,3,2	23) 64,63,2,1	44) 39,40,1,1	65) 6,22,4,3
3) 3,4,3,2	24) 63,62,2,1	45) 40,41,1,1	66) 22,23,4,3
4) 4,5,3,2	25) 62,61,1,1	46) 41,42,1,1	67) 23,24,4,3
5) 5,6,3,2	26) 61,60,1,1	47) 42,30,1,1	68) 11,25,4,3
6) 6,7,3,2	27) 60,59,1,1	48) 30,43,1,1	69) 25,26,4,3
7) 7,8,3,2	28) 59,58,1,1	49) 43,44,1,1	70) 26,27,4,3
8) 8,9,3,2	29) 58,57,1,1	50) 44,45,1,1	71) 16,28,4,3
9) 9,10,3,2	30) 57,56,1,1	51) 45,46,1,1	72) 28,29,4,3
10) 10,11,3,2	31) 56,55,1,1	52) 46,21,1,1	73) 29,30,4,3
11) 11,12,3,2	32) 55,1,1,1	53) 2,47,1,1	
12) 12,13,3,2	33) 1,31,1,1	54) 47,48,1,1	
13) 13,14,3,2	34) 31,32,1,1	55) 48,49,1,1	
14) 14,15,3,2	35) 32,33,1,1	56) 49,50,1,1	
15) 15,16,3,2	36) 33,34,1,1	57) 50,51,1,1	
16) 16,17,3,2	37) 34,24,1,1	58) 51,52,1,1	
17) 17,18,3,2	38) 24,35,1,1	59) 52,53,1,1	
18) 18,19,3,2	39) 35,36,1,1	60) 53,54,1,1	
19) 19,20,3,2	40) 36,37,1,1	61) 54,67,2,1	
20) 20,21,3,2	41) 37,38,1,1	62) 67,68,2,1	
21) 66,65,2,1	42) 38,27,1,1	63) 68,69,2,1	

APPENDIX 7

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Machine Design
Bridge Crane Term Project

Richard Sciscente

*79897

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BRIDGE CRANE DATA

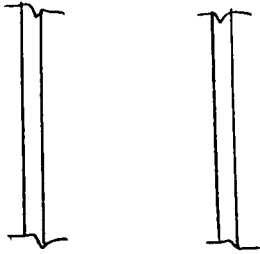
Birthday = May 2, 1967

span = $9 + \text{\#birthmonth} = 14$ meters

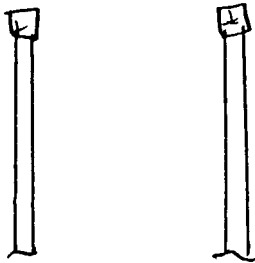
load = $27 + \text{\#birthday} = 29$ metric tons

The purpose of this term project is to design a bridge crane for the surface of the moon that has a span of 14 meters, a height of 15 meters, and a length of 30 meters. The crane must carry a maximum load of 29 metric tons.

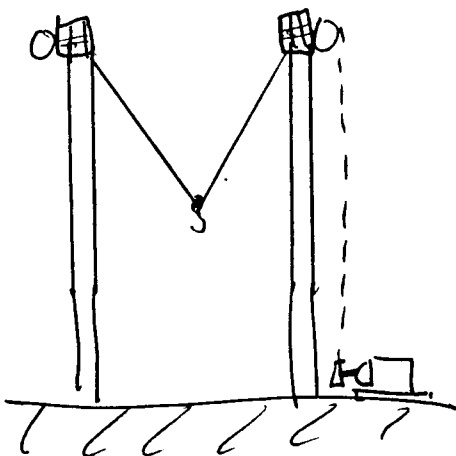
CRANE DESIGN



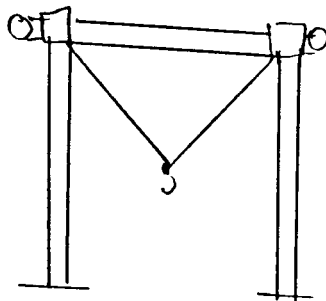
The design begins with the side supports. There will be two columns on each side.



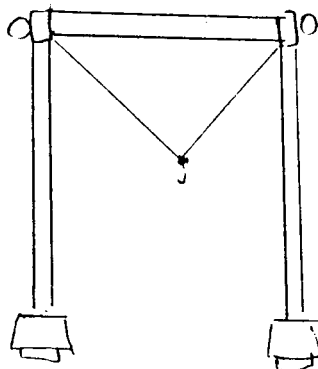
The columns will meet and be joined together at the top by a corner.



Attached to each corner is an electric motor. These motors control the lengths of cable that hang from each corner. The ends of the cables are joined together. The load is applied at that point.



The load will force the columns inward. There is a crosspiece between the corners that is in compression that will hold the corners apart.



The bottoms of the columns are set into a base on each side. The bases have four wide wheels each. Each base is driven by a separate motor.

THE CABLE

steel cable 6x19 $E = 12,000,000$ psi

Improved plow steel $S_{ult} = 200,000$ psi

density = 0.000 078 N/(cubic mm)

computed for long life and continual use

from fig. 12.23 $2T / (S_{ult} d_r d_s) = .0014$

$d_r = 16 d_w$

$d_s = 24 d_r$

WEIGHT CONTRIBUTION OF CABLE TO THE LOAD

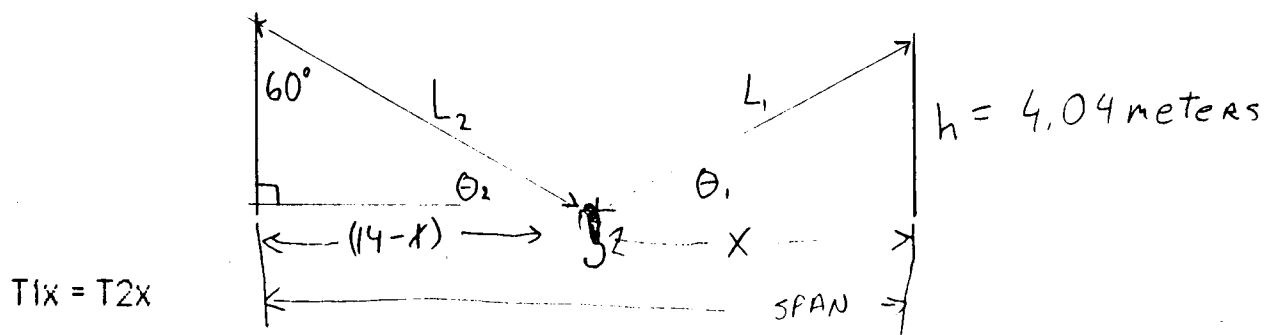
$$W_c = .286 \frac{\text{lbs}}{\text{in}^3} \left(\frac{(\text{lbs at the moon})}{6 \text{ lbs earth}} \right) (6 \times 19) \left(\frac{\pi}{4} d_w^2 \right) (\text{LENGTH EXTENDED})$$

$$W_c = 0.0166713 (d_r)^2 L$$

$$(d_r)^2 = \frac{2(T_{\text{ension}})}{24(.0014) 200,000 \text{ psi}} = (T > 2.976 \times 10^{-4}) \text{ in}^2$$

$$W_c = 4.9617 \times 10^{-6} (T L)$$

For simplicity, the cables will be connected at 19.04 meters above the ground. This is so the angle formed by the column and the cable when the load is at top center will be 60 degrees. That will make the tension in each cable equal to the weight of the load. The weight of the cable is negligible compared to the massive load. Any uncertainties are negated by a FS of 4.



$$T_1 x = T_2 x$$

$$T_1 y + T_2 y = \text{load}$$

$$T_1 \cos \theta_1 = T_2 \cos \theta_2$$

$$T_2 = \frac{T_1 \cos \theta_1}{\cos \theta_2}$$

$$T_1 \sin \theta_1 + T_2 \sin \theta_2 = \text{LOAD} = T_1 [\sin \theta_1 + \cos \theta_1 \tan \theta_2]$$

$$T_1 = \frac{\text{LOAD}}{\sin \theta_1 + \cos \theta_1 \tan \theta_2}$$

$$= \frac{\text{LOAD}}{\frac{h}{\sqrt{h^2 + x^2}} + \left(\frac{x}{\sqrt{h^2 + x^2}} \frac{h}{\text{SPAN} - x} \right)}$$

$$S = \text{SPAN}$$

$$= \frac{\text{LOAD}}{h} \left(\frac{\sqrt{h^2 + x^2}}{1 + \frac{x}{S - x}} \right)$$

$$T_1 = \frac{\text{LOAD}}{h \cdot S} (\sqrt{h^2 + X^2} \cdot S - X)$$

$$\text{SET } \frac{dT_1}{dX} = 0$$

$$\frac{dT_1}{dX} = \frac{\text{LOAD}}{h \cdot S} \left[\sqrt{h^2 + X^2} (-1) + (S - X) \frac{1}{2} \frac{2X}{\sqrt{h^2 + X^2}} \right]$$

$$\sqrt{h^2 + X^2} = \frac{(S - X)X}{\sqrt{h^2 + X^2}}$$

$$h^2 + X^2 = SX - X^2$$

$$2X^2 - SX + h^2 = 0$$

$$X^2 - \frac{S}{2}X + \frac{h^2}{2} = 0$$

$$X = \frac{\frac{S}{2} \pm \sqrt{\frac{S^2}{4} - 2h^2}}{2}$$

$$X = \frac{7 \pm \sqrt{49 - 2 \cdot 16 \cdot 33}}{2}$$

$$X = \frac{7 \pm 4.04145}{2}$$

$$X = \frac{7 \pm h}{2}$$

$$X = 1.48 \text{ meters}, 5.5207 \text{ meters}$$

$$T_{\text{max across the top}} = 1.02525 (\text{LOAD})$$

$$W_{\text{cable}} = W_c = [1.02525 (\text{LOAD}) + W_c] 4.9617 \times 10^{-6} \cdot (\text{FS})$$

$$\text{FS} = 4$$

$$W_c = \frac{(1.02525)(4.9617 \times 10^{-6}) \left[29000 \text{ Kg} \left(\frac{9.8 \text{ m/s}^2}{6} \right) \right] 4}{1 - [4 \times 4.9617 \times 10^{-6}]}$$

$$= \frac{.9638}{.99998} \frac{\text{N}}{\text{in}}$$

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$$= .9638 (1.00001985)$$

CONTRIBUTION FROM THE MASS OF CABLE

$$= .9638 \frac{\text{NEWTONS}}{\text{in}}$$

$$\text{MASS OF CABLE} = \frac{.9638 \text{ N/in} \cdot 6}{9.8 \text{ m/s}^2} = .5901 \frac{\text{Kg}}{\text{in}}$$

THE MASS OF THE CABLE IS NEGLIGIBLE, SO THE WORST CASE WAS ACROSS THE TOP. SPACE LIMITATIONS KEEP THE CABLE FROM HANGING DOWN TO A STEEPER MORE OPTIMUM ANGLE. THE STEEPER ANGLE WOULD MEAN TALLER SUPPORT COLUMNS. MINE ARE ALREADY DANGEROUSLY HIGH

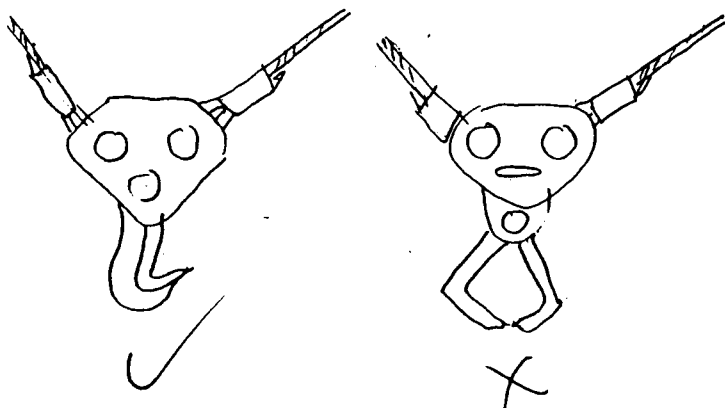
$$\begin{aligned} \text{TOTAL MASS OF CABLE} &= 2 \cdot \sqrt{5^2 + (15+4)^2} \cdot .5901 \frac{\text{Kg}}{\text{in}} \cdot \frac{1 \text{ in}}{.0254 \text{ m}} \\ &= 1098.1 \text{ Kg OF CABLE} \end{aligned}$$

0.225 in = dw FOR the 6x19 steel cable

THE HOOK

Look in handbook where these people have summarized their studies

The hook should be designed by someone who knows what kind of hook is needed. The estimated mass is 100 kg.



THE CROSSPIECE

The crosspiece is going to be a slender compression member. Its maximum load will be the maximum horizontal component of cable tension. This will be the tension when the load is at top center. The crosspiece will be designed with fixed ends. Since buckling can occur in any direction, the crosspiece will be a hollow tube to maximize I in all directions. The material that I would like to use for this type of piece is the material that the space shuttle's main tanks are made of. It would be plentiful enough if it set into an orbit around the earth before being discarded. It could then be salvaged, but that is someone else's project. I will settle for the aluminum described in Table 2-1 of our text.

143

$$P_{MAX} = \frac{4\pi^2 EI}{L^2}$$

$$T_{XMAX} = P_{MAX} = \frac{29000 \text{ Kg}}{2 \cdot \tan 30^\circ} \left(\frac{9.8 \text{ m/s}^2}{6} \right) = 41020 \text{ N}$$

$$\frac{4\pi^2 EI}{L^2} = 41020 \text{ N (F.S.)}$$

$$I = \frac{41020 \text{ N} (4) (14 \text{ m})^2}{4\pi^2 \frac{69000000000 \text{ N}}{\text{m}^2}} = .000011806 \text{ m}^4 = \frac{\pi}{64} (OD^4 - ID^4)$$

$$OD^4 - ID^4 = .0002405$$

$$\frac{OD}{OD-ID} < 120 \text{ to prevent crippling}$$

$$OD = .29 \text{ meters}$$

$$ID = (.29^4 - .0002405)^{1/4} = .2875 \text{ meters}$$

$$\frac{OD}{OD-ID} = 116.12 < 120$$

TRIAL AND ERROR

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Threaded Ends of Crosspiece

$$\text{pitch} = p = 0.125 \text{ in} = 0.003175 \text{ meter}$$

$$\text{height of thread} = .6495p = 0.002062 \text{ meter}$$

$$\text{pitch diameter} = .122 \text{ meters}$$

$$\text{root diameter} = .120 \text{ meters}$$

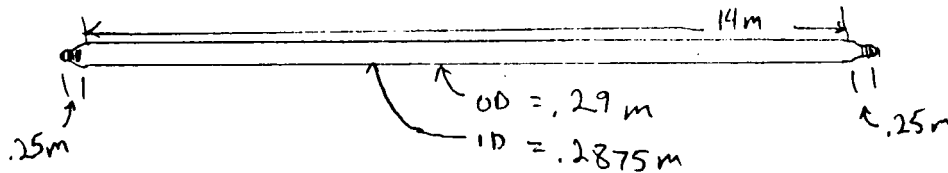
$$\text{helix angle} = \left(\frac{0.003175}{\pi \cdot .122} \right) = \tan^{-1} (.00856)$$

$$\alpha = .49^\circ$$

$$\text{Weight} = .0000271 \frac{N}{(mm)^3} \left[14000 \text{ mm} \frac{\pi}{4} [(290)^2 - (287.5)^2] (mm)^2 + \frac{\pi}{4} (124 \text{ mm})^2 250 \text{ mm} \right]$$

$$W = .0000271 \frac{\pi}{4} [20212500 + 3844600]$$

$$W = 512.025 \text{ N}$$

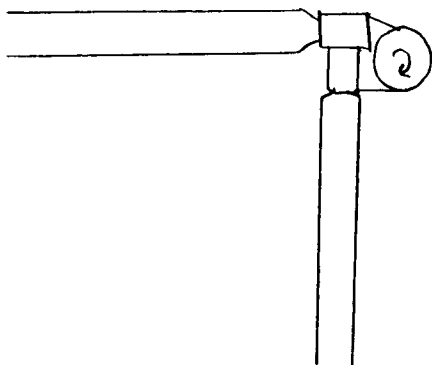


The shape of the cross piece is not purely straight. If it were, it would sag when simply supported. Then it would buckle easier. The cross piece must be shaped so that when it is simply supported, it will sag and then be perfectly straight.

$$\theta = \frac{w l^3}{24EI} = .0137^\circ$$

very risky !!
then subject to
oscillations!

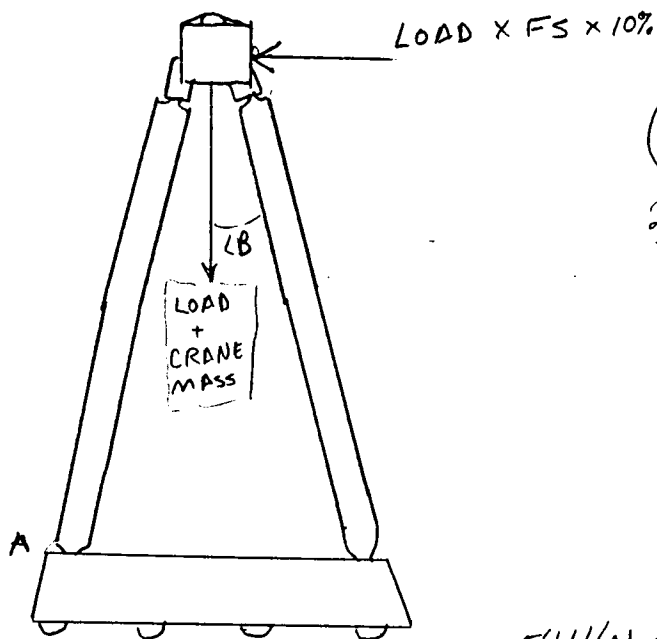
COLUMNS



There are two columns on each side.

$\angle B$ is such that (load \times FS \times 10%) of horizontal force at the top will just put one of the columns in tension.

$$F_x = \text{LOAD} \times \text{FS} \times 10\% \\ = \frac{29000 \times 4 \times 9.8}{60} = 18950 \text{ N}$$



$$\sum M_A = 0$$

$$(18950 \text{ N}) (19.04 \text{ m}) = 19.04 \text{ m} \tan B (g) (29000 + \text{MASS OF CRANE})$$

$$\frac{29000 \cdot 4}{10} = \tan B \cdot 29000 + 2000 \text{ (MAX)}$$

$$\frac{11.6}{31} = .3742 = \tan B$$

$$\angle B = 20.5^\circ$$

$$F_{\text{COMPRESSION}} = P_{\text{MAX}} = \frac{18950 \text{ N}}{\sin 20.5^\circ} = 54111 \text{ N}$$

$$54111 \text{ N} = \frac{4 \pi^2 EI}{L^2}$$

$$I = \frac{54111 (19.04)^2}{4 \pi^2 (\cos 20.5^\circ)} \cdot \frac{1}{69000,000,000} \text{ m}^4 \\ = 0.000008207$$

$$OD^4 - ID^4 = .000167$$

$$ID = (OD^4 - .000167)^{\frac{1}{4}}$$

$$OD = .27$$

$$ID = .26785$$

$$10$$

$$\frac{OD}{OD-ID} < 120$$

Threaded Ends of columns

$$p = 0.003175$$

$$h = 0.002062$$

$$pd = .112 \text{ m}$$

$$rd = .110 \text{ m}$$

$$\text{helix angle} = \arctan\left(\frac{.003175}{\pi .112}\right) = .00905$$

$$\alpha = .518$$

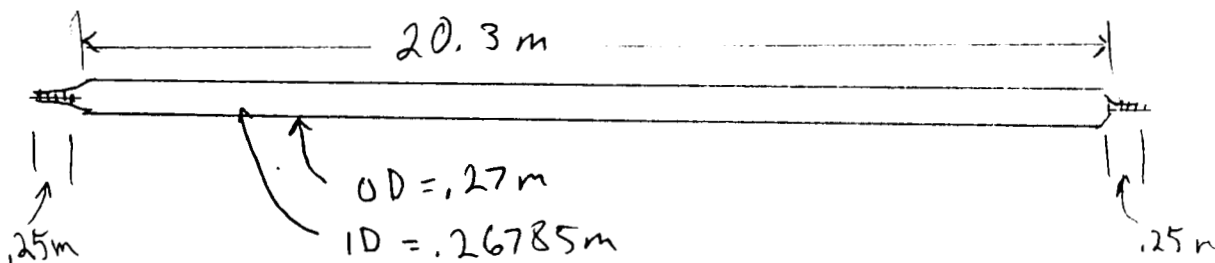
Weight

$$W = \left[\left(\frac{19.04 \text{ m}}{\cos 20.5} \right) \frac{1000 \text{ mm}}{\text{mm}} (270^2 - 267.85^2) + (250 \text{ mm} (112 \text{ mm}^3)) \right]$$

$$\times (.0000271) \frac{\pi}{4} \frac{N}{(\text{mm})^3}$$

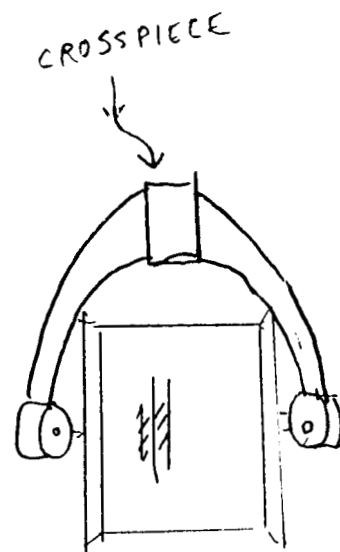
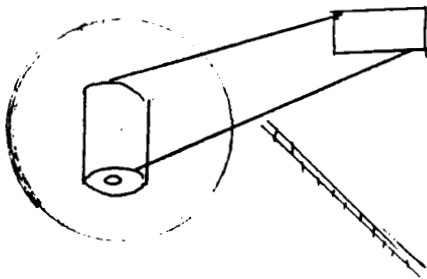
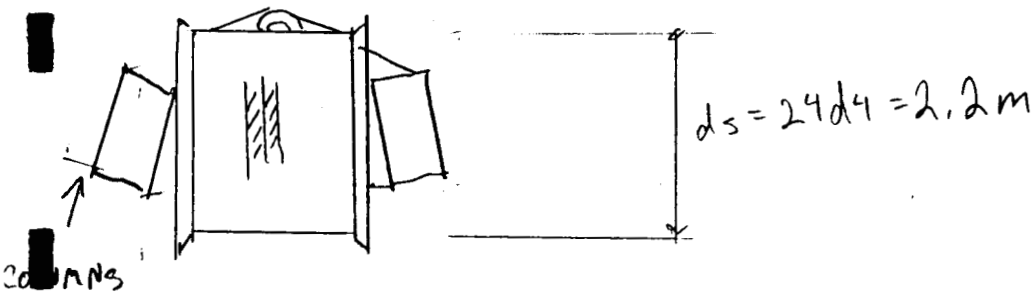
$$W = [23506012 + 3136000] .0000271 \frac{\pi}{4} N$$

$$W = 567 \text{ N}$$



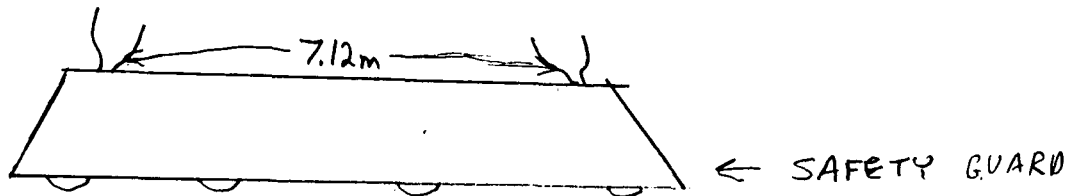
THE CORNERS

The corner is the piece that holds the crane together. It is very important to my future in the space industry that this piece does not fail. The job that the corner is required to do gives it an odd shape. To avoid any damage due to stress concentrations in this odd shape, this piece must be severely overdesigned. It should be durable to the point of indestructable. It is easier to overdesign the corners. They don't weigh that much anyway.



ESTIMATED MASS 870 Kg each

THE BASES



Each base consists of a frame to hold the column bases and distribute the loads of the columns to the four wide wheels of the base.

IT MUST BE LIKE A CART TO HOLD THE CRANE'S WEIGHT APPLIED BY THE COLUMNS AND HAVE THE STRENGTH IN TENSION TO HOLD THE BOTTOM OF THE COLUMNS TOGETHER. EACH BASE MUST BE DRIVEN SEPARATELY BY A MOTOR. A PROBABLE ESTIMATION OF THE MASSES OF A COMPLETED BASE COULD BE 2000 Kg.

CONCLUSION

THIS CRANE IS DESIGN TO BE WEIGHT EFFECTIVE.
 A SECONDARY CONSIDERATION IS THAT IT CAN BE
 ASSEMBLED ON THE MOON'S SURFACE. IN CALCULATING
 THE DESIGN CRITERIA FOR THOSE PARTS OF THE
 CRANE THAT I FELT I WAS NOT QUALIFIED
 TO GIVE AN EXACT ANSWER, I DID MY BEST
 TO PROPOSE LOGICAL ESTIMATION

$$\text{SPAN} = 14 \text{ METERS}$$

$$\text{HEIGHT} = 15 \text{ METERS}$$

$$\text{MAX LOAD} = 29 \text{ METRIC TONS}$$

$$\begin{aligned} \text{MASS} = & M_{\text{CABLE}} + M_{\text{CROSSPIECE}} + M_{\text{HOOK}} + 2M_{\text{CORNER}} \\ & + 4M_{\text{COLUMN}} + 2M_{\text{BASE}} \end{aligned}$$

$$\begin{aligned} \text{MASS} = & 1098.1 \text{ kg} + \frac{512.025}{9.8} \text{ kg} + 100 \text{ kg} + 2 \cdot 800 \text{ kg} \\ & + 4 \cdot \frac{567}{9.8} \text{ kg} + 4000 \text{ kg} \end{aligned}$$

$$\text{MASS} = 7080 \text{ kg}$$

$$PI = \frac{29000 \text{ kg} (14 \text{ m})^2}{7080} = 802.8$$

$$PI = 802$$

APPENDIX 8

A LUNAR-BASED PROPULSION SYSTEM
FOR A LAUNCH AND LANDING FACILITY

Preliminary Design Proposal

Submitted to

Dr. Jerald N. Linsley
Principal Investigator

and

Mr. Eric B. Jenson
Project T.A.

NASA/USRA/UADP

Florida Institute of Technology
150 W. University Blvd.
Melbourne, Florida 32901
(305)-768-7326

12/03/87

for

CHE 4095
Chemical Engineering Senior Design Project

and

EGN 4001
Special Projects in Space Systems Design - I

Report by

Mark Ventura

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ABSTRACT

A preliminary lunar-based propulsion system design is proposed to support a lunar launch and landing facility. A pumped hydrogen/oxygen (H/O) propulsion system supports the "baseline transportation fleet." Oxygen will be supplied by a lunar-based oxygen production facility, while hydrogen must be imported from Earth. Propellant requirements and fueling operations are analyzed. Safety considerations accompany the propulsion system design for the launch and landing facility. Future work is required to complete the design of the lunar-based propulsion system.

INTRODUCTION

A lunar-based propulsion system will need to accompany the design of a lunar launch and landing facility for future support of lunar missions. This design study is being conducted at the Florida Institute of Technology (FIT) as part of the University Advanced Design Project (UADP). Support is recieved from the National Aeronautics and Space Administration (NASA). My design of the lunar-based propulsion system is being conducted for the course CHE-4095, Chemical Engineering Senior Design Project I, taken in conjunction with EGN-4001, Special Projects in Space Systems Design - I.

The existence of a lunar base will be largely dependent on the transportation system that supports it. A lunar-derived propellant system could provide the most important resource for the transportation infrastructure. The most important characteristic of an efficient lunar base propulsion system is the degree of lunar self-sufficiency and adequate propulsion system performance.

A pump-fed hydrogen/oxygen (H/O) propulsion system has been selected for the lunar-based propulsion system. Primary reasons for this choice include that this system is already state-of-the-art, hydrogen offers excellent performance with oxygen, and that this system has an estimated high delivered specific impulse (1).

Oxygen, being the most abundant element on the moon,

accounts for over 40% of the lunar composition (2). Thus, the utilization of lunar oxygen to serve as a bipropellant for the propulsion system is an obvious possibility. We assume an operational lunar oxygen production facility will supply the propellant. Unfortunately, the moon does not possess an adequate supply of hydrogen. In fact, only trace amounts of hydrogen existing in concentrations of only several part-per-million could be found. For this reason hydrogen must be imported from Earth.

It must be understood that the liquid hydrogen and oxygen used in this lunar-based propulsion system are cryogenic fluids. A cryogenic fluid can only remain in the liquid phase at extremely low temperatures (3). Some of the physical properties of these liquids can be found in Table I. These fluids are cold enough to freeze human tissues. Their expansion upon vaporization can create high pressures, capable of breaking containers. Also, cryogenic hydrogen is a fuel; it can burn and explode.

Heat leaks into a cryogenic system rather than out. This must be offset by adding refrigeration, which is usually costly. To hold this cost to a minimum, low-temperature vessels are usually compacted to reduce exposed surface areas and increase insulation. Because of this, it is often difficult to isolate and detect leaks, even though every precaution is taken to design and build to the highest possible standards. When leaks do occur, the materials tend to diffuse rapidly through the bulk of

the insulation. Thus combustion or the loss of valuable propellant resources may result.

The technical character of the hazards which exist for the lunar-based propulsion system can be separated into four categories: (1) Loss of cryogenics resulting from boiloff; (2) Brittleness of structural materials at low temperatures; (3) High pressure arising from confinement of liquids; and (4) Hazards due to flammability.

PROPELLANT REQUIREMENTS

Before a lunar-based propulsion system can be adequately designed, a propellant requirement estimate needs to be determined. The amount of cryogenic bipropellants needed to support the "baseline transportation fleet" is a critical factor in the propulsion system design.

NASA/JSC supplied the mission model, which is shown in Figure I and Table II (2). The nominal mission model covers 20 years from 1995 to 2015. The darkened bars in Figure I represents the initial mass to the lunar surface or lunar orbit. The open bars denote payload delivery to the lunar surface/orbit not directly associated with propellant production.

The propellant requirements for the lunar-based propulsion system are depicted in Figure II. These various requirements depend on the concept chosen to represent the "baseline transportation fleet." These concepts are displayed in Figure III. For our study purposes, we will chose Concept I, H/O OTV and Lander. We will also assume a specific impulse of 470 seconds for the OTV and Lander, a 15.9 MT maximum payload, an oxidizer/fuel (O/F) ratio of 5.5, no aerobrake technology available, and an adequate supply of lunar oxygen to furnish the OTV and Lander for the return trip to Lower Earth Orbit (LEO). Appendix I lists the engine/performance data for this concept.

The storage capacity of the lunar oxygen supply should be

able to furnish two fleets. Since each OTV and Lander mission from Lunar Surface Base (LSB) to LEO requires 56.3 MT of oxidizer, approximately 115 MT of cryogenic oxygen should be stored on the lunar surface.

A lunar-based hydrogen supply should be able to furnish two missions also. This supply will be imported from Earth. The cryogenic hydrogen will be defueled from the lander and stored in the propulsion system. This fuel will be used for emergency use only.

PROPULSION SYSTEM DESIGN

Now that the propellant requirements have been estimated for the lunar-based propulsion system, the propulsion system design can be integrated. A similiar tank, pump, and piping arrangement will be used for both cryogenes (see Figure IV). Although both systems require proper insulation to prevent boiloff, the liquid hydrogen system should have excessive insulation since its loss would pose serious problems. The major components of these systems include storage tanks, suction and discharge lines (piping), relief valves, back pressure valves, centrifugal pumps, strainers, shut-off valves, drainage valves, and check valves (4).

The rules for designing storage vessels in use at cryogenic temperatures and pressures are quite well established, at least for an Earth environment. The A.S.M.E. code is very specific in describing procedures to be used in determining stress values, material thickness, design details, material selections, etc. However, lunar considerations, such as an ambient atmospheric pressure, need to be accounted for.

Suction lines for the propulsion system should be short. Long lines may result in poor performance notably underfeeding, nonlinearity, and vibration or shaking of piping. Heavy insulation is required for the suction lines.

The discharge lines should also be as short and as free from

bends as possible. Long lines create higher pressure drops and subject to mass inertia effects resulting in hydraulic shock.

A process line relief valve is required for system protection and should be installed in the discharge line close to the pump. This valve will protect the line from damage due to plugging or accidental valve closure.

A back pressure valve should not be used to prevent a positive liquid level from draining or siphoning through the pump to an atmospheric discharge. The sole purpose of the back pressure valve is to linearize pump delivery, not to confine the contents of the supply tank. As the valve wears, some minor leakage may occur. Monitoring this valve for such defects would be necessary.

The centrifugal pump should be located as close to the cryogenic supply tank as possible to maximize efficiency. This common pump has a very large pumping capacity. However, blockage or breakage can result of faulty operation.

A strainer should be employed between the storage tank and the centrifugal pump. This mechanism would prevent foreign matter from entering the pump with consequent possibility of interfering with check valve operation.

Strategically located shutoff and check valves should be

incorporated to permit servicing the pump without draining the entire system. Drain valves should be installed at the lowest point in the discharge line.

The lunar-based propulsion system will be located adjacent to the service and control facility (see Figure V). This option will allow efficient fueling and defueling of the lunar lander during service procedures. Also, this will keep the bipropellant storage supply away from the launchpad, minimizing the possibility of combustion. The option of transporting the bipropellants from a safe distance to the launchpad via piping is irrational, since discharge lines should be as short as possible.

Protection measures need to be taken to protect the lunar-based propulsion system from the hazards of the lunar environment. Such hazards include solar radiation, micrometeorite bombardment, and extreme temperature changes. By burying the system ten feet in the lunar surface, these hazardous environmental effects will be minimized.

SAFETY CONSIDERATIONS

Foremost in our criteria for the design of the lunar-based propulsion system is the development of a safe process to perform the functions required, such as fueling the space vehicles. The choice of operating conditions and process equipment for safely carrying out the process is integral. It is very important that no hazardous conditions exist with the propulsion system. Therefore we will apply solubility, vapor pressure, adsorptive capacity, and equilibria conditions to avoid hazardous conditions. Also, automating the propulsion system process as much as possible will certainly aid in minimizing possible hazardous conditions.

We apply safety concepts in both the procurement and operation of the mechanical equipment used in our lunar-based propulsion system. This equipment includes compressors, expansion machines, pumps, and mechanical refrigerators. The equipment operates over a wide range of pressures and temperatures, and handles a variety of fluid mixture ratios.

All moving equipment must be protected against hazards. The hazards may be classified as mechanical, material compatibility, equipment malfunction, and faulty operation, which involves the human element. Again, automation through the use of robotics must again be emphasized for safe propulsion system operations.

Sealing is a major problem which may develop in the lunar-

based propulsion system. Oxygen and hydrogen fluids may create fire hazards. Either of the cryogenic fluids may cause pressure ruptures if they leak into tight enclosures which are not constructed to contain pressures and are then allowed to warm up. Direct liquid leaks on surrounding structures which are highly stressed can cause dangerous equipment or structural failures.

If cryogenic oxygen is lost through leakage, the lunar oxygen production facility could replenish the supply. However, since the hydrogen fuel is imported from Earth, the loss of this cryogen could be detrimental to the survival of the lunar base. Thus, heavy insulation is required to protect and retain the valuable hydrogen fuel.

Particular consideration must be given to liquid hydrogen. Reviewing the combustion properties of hydrogen, we may note that the hydrogen flammable limits are easier to obtain, ignition requires lower energies, flames propagate faster and are more difficult to extinguish (5). Compared to household fuels, hydrogen demonstrates some additional degree of hazard in these respects.

Three major circumstances or conditions of hazard can exist in conjunction with liquid hydrogen in the lunar-based propulsion system. The first of these is the presence of oxidants in the liquid hydrogen. A second potential hazard, perhaps the major one, is that of spilling. A third hazard, and perhaps the most

frequent one in modern Earth-based systems, is that of vented gas.

In the summarizing the degree of hazard, we may note that the neccessity of storing and transferring liquid hydrogen in closed, well-insulated equipment protected with relief devices almost eliminates the possibility of boiloff and puts the pressure rupture problem on the same plane as other pressure applications. The only difference in degree is in the dissipation and combustion properties.

Three conditions must be met simultaneously in order to have combustion anywhere in the lunar-based propulsion system. First, there must be a fuel. Second, there must be an oxidant. Third, there must be a source of ignition.

Possible sources of ignition include impact, heat produced by dynamic effects, heat produced by friction, heat produced by chemical reaction or decomposition, or static electricity or other electric spark.

FUELING AND DEFUELING OPERATIONS

The requirements of the lunar-based propulsion system are to supply lunar-derived liquid oxygen for the OTV and Lander's return to trip to LEO and to store an emergency supply of liquid hydrogen fuel. Cryogenic hydrogen used by the OTV and Lander for the entire mission will be Earth-derived.

Oxidant will be transported into the Lander, while a certain amount of hydrogen will be defueled and placed into emergency storage. Automated mechanisms should carry out the fueling and defueling operations for reasons of efficiency and safety.

The general rules for fueling and defueling liquid hydrogen and oxygen in the lunar-based propulsion system are: (1) Provide good insulation for the system; (2) Monitor the system for oxidants and monitor the surroundings for hydrogen; (3) Make all transfers in closed, carefully prepared systems; (4) Use isolated electrical equipment; (5) Provide protection against static buildup and sparks; and (6) Allow no open flame.

CONCLUSIONS AND RECCOMENDATIONS

The efficiency, capability, and evolution of a lunar launch and landing facility will be largely dependent on the propulsion system that supports it. Most of the technology required to build such a lunar-based propulsion system is state-of-the-art. Other than space basing, no new technologies are needed for the hydrogen/oxygen systems. However new technology options should be explored.

The completed design of a pump-fed hydrogen/oxygen propulsion system for a lunar launch and landing facility will require much more work. All of the materials and dimensions of the system's components need to be determined. Also, the suction pressures, head losses, static siphoning, and suction lifts generated by the propulsion system need to be calculated. Once these parameters are established, applications of solubility, vapor pressure, and equilibria conditions will be required to minimize hazardous conditions. A substantial amount of robotics also needs to be integrated into the system. Finally, a design of a lunar-based propulsion system to support a lunar launch and landing facility can be completed.

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APPENDIX I
ENGINE/FUEL DATA FOR PROPULSION SYSTEM CONCEPT

H/O OTV & LANDER
SPECIFIC IMPULSE = 470 sec
O/F RATIO = 5.5
LLOX AVAILABLE
15.9 MT PAYLOAD
NO AEROBRAKE

This is a two vehicle configuration which uses lunar propellants. The OTV travels to LLO carrying a payload and propellant for the Lander. The Lander makes 3 round-trips from LSB to LLO. It carries the OTV payload to LSB and delivers lunar propellant to the OTV. After 3 Lander trips, the OTV departs for LEO, loaded with lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:	26649.91 kg
LUNAR LOX USED BY OTV:	16036.96 kg
LUNAR FUEL USED BY OTV:	0 kg
LUNAR LOX RETURNED:	10612.85 kg
LEO-BASED LOX BURNED:	35436.17 kg

OTV DESIGN

OTV ENGINE DATA

Specific Impulse:	470 seconds
Number of Engines:	2
Thrust per Engine (N):	33361

Mass of each engine and its thrust structure: 95 kg

O/F Ratio (O/H): 5.5

OTV MASS (kg):

Dry Mass: 1030

Aerobrake Mass 0

LOX Tank Mass 141.7447

Fuel Tank Mass 1580.146

Pressure Tank Mass 0

Total Mass: 2751.89

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 35436.17

LOX Carried for OTV: 35436.17

LOX Carried for Lander: 0

Additional LOX Storage Capability for Return Trip: 0

Fuel Capacity for OTV: 9358.751

Total Propellant Capacity: 44794.92

Percent of Return Trip LOX from LSB: 100

Percent of Return Trip Fuel from LSB: 0

Payload to LSB: 15673

Return Payload Capability: 15673

Mass Fraction: .9421224

LANDER DESIGN

LANDER ENGINE DATA:

Specific Impulse: 470 seconds

Number of engines: 2

Thrust per engine (N): 33361

Mass if each engine and its thrust structure (kg): 95

O/F Ratio (O/H): 5.5

LANDER MASS (kg):

Dry Mass: 1030

Landing Gear Mass: 1846.201

LOX Tank Mass: 83.4128

Fuel Tank Mass: 367.7746

Pressure Tank Mass: 0

Total Mass: 3327.588

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20853.2

Fuel Capacity: 3791.49

Total Propellant Capacity: 24644.69

Percent of Lander LOX supplied from LSB: 100

Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873

Liftoff Payload: 8973

Tank Structure for Refueling OTV: 89.72999

Mass Fraction: .8810461

TABLES

TABLE I

SOME PHYSICAL PROPERTIES OF HYDROGEN, PROPANE, OCTANE, OXYGEN, AND AIR

	H ₂	C ₃ H ₈	C ₈ H ₁₈	O ₂	AIR
Boiling Point, °F	-422.99	-43.7	+258.2	-297.3	-317.9
Liquid Density, lb/ft ³ @ B. P.	4.43	36.2	44	71.2	54.37
Latent Heat, BTU/lb-mole	389	8076	16,620	2932	2556
Gas Density, lb/ft ³ at 70°F, 1 ATM.	.00522	0.1154	—	.0628	.07493
Liquid to Gas Expansion Ratio, B. P. to 70°F	851.33	—	—	862	726.13
Diffusion Coefficients (into air) gm/cm ² /sec	.68	—	—	.17	—

TABLE II JSC LSB MISSION MODEL, JANUARY 1986

YEAR	DESTINATION			NO OF PAYLOADS	MASS, MT (lbs)	MANNED
	LO	LS	RETURN			
1993	X			1	22.7 (50)	
1996	X			1	22.7 (50)	
1999		X		1	8.2 (18)	
2003			X	1		X
2004			X	1		X
2005		X		2	2.3 (5)	X
		X		3	19.2 (42.4)	
2006		X		1	19.2 (42.4)	
				3	2.3 (5)	X
2007	X			1	22.7 (50)	
		X		3	2.3 (5)	X
		X		1	19.2 (42.4)	
2009		X		9	2.3 (5)	X
		X		2	19.2 (42.4)	
		X		3	15.9 (35)	
2010		X		4	15.9 (35)	
		X		3	10.4 (23)	X
		X		3	3.6 (8)	X
	X			1	22.7 (50)	
2011		X		1	15.9 (35)	
		X		2	10.4 (23)	X
		X		4	3.6 (8)	X
2013		X		5	15.9 (35)	
		X		4	10.4 (23)	X
		X		1	3.6 (8)	X
2014		X		3	10.4 (23)	X
		X		3	3.6 (8)	X
2015		X		1	3.6 (8)	X
		X		1	45.4 (100)	
		X		1	103.9 (240)	

FIGURES

LSB MISSION MODEL

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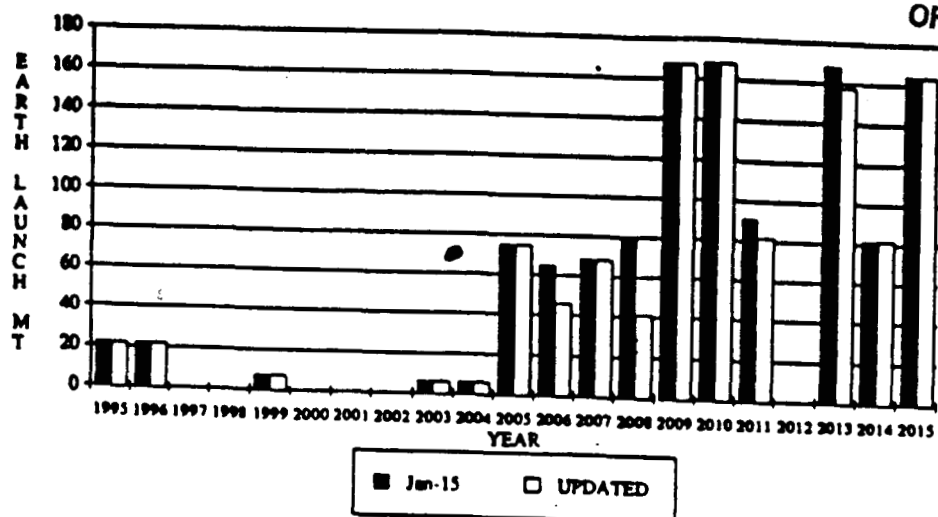


FIGURE I JSC MISSION MODEL, JANUARY 1986

- (0) BASELINE, H/O OTV & LANDER (NO LUNAR PROPELLANT)
- (1) H/O OTV & LANDER (LLOX AVAILABLE), DRY WEIGHT REDUCTION
- (2) H/O OTV & LANDER (LLOX & LH₂ AVAILABLE), DRY WEIGHT REDUCTION
- (3) H/O OTV & AV LOX LANDER; LANDER Isp = 260, O/F = 2.1
- (4) H/O OTV & LANDER; OF = 8.7 (Isp = 421)
- (5) H/O OTV & LANDER; OF = 10.6 (Isp = 384)
- (6) H/O OTV & LANDER; I_{sp} = 460
- (7) H/O OTV & LANDER; I_{sp} = 490
- (8) H/O OTV & LANDER; PAYLOAD = 10MT
- (9) H/O OTV & LANDER; PAYLOAD = 20MT
- (10) H/O OTV & LANDER; NO AERO BRAKE
- (11) H/O OTV & LANDER; AEROBRAKE MASS = 18% OF REENTRY MASS
- (12) H/O OTV & LANDER; AEROBRAKE MASS = 20%
- (13) H/O OTV & LANDER; AEROBRAKE MASS = 25%
- (14) H/O OTV & LANDER; AEROBRAKE MASS = 30%
- (15) SiH₄ OTV & LANDER
- (16) Al-H/LOX OTV & LANDER
- (17) H/O OTV & LANDER WITH LLOX RETURN TO LEO

FIGURE II , LEGEND FOR VEHICLE SUMMARY

PROPELLANT REQUIREMENTS SUMMARY
(LUNAR PROPELLANTS AVAILABLE IN 1995)

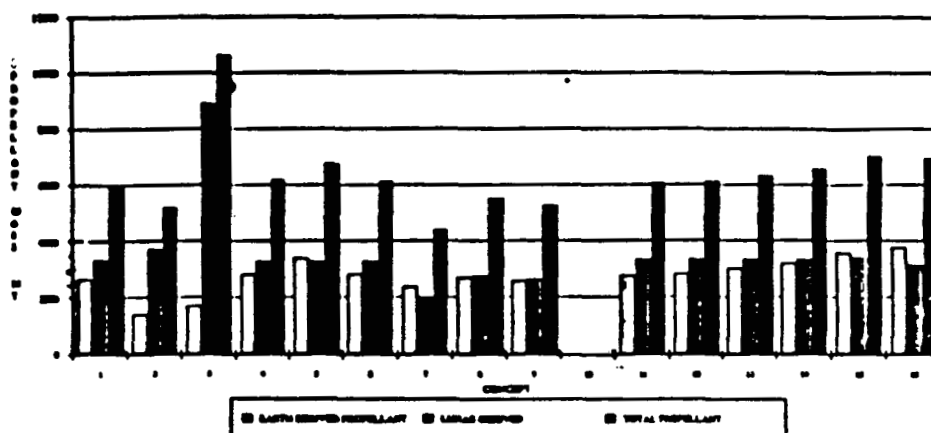


FIGURE III

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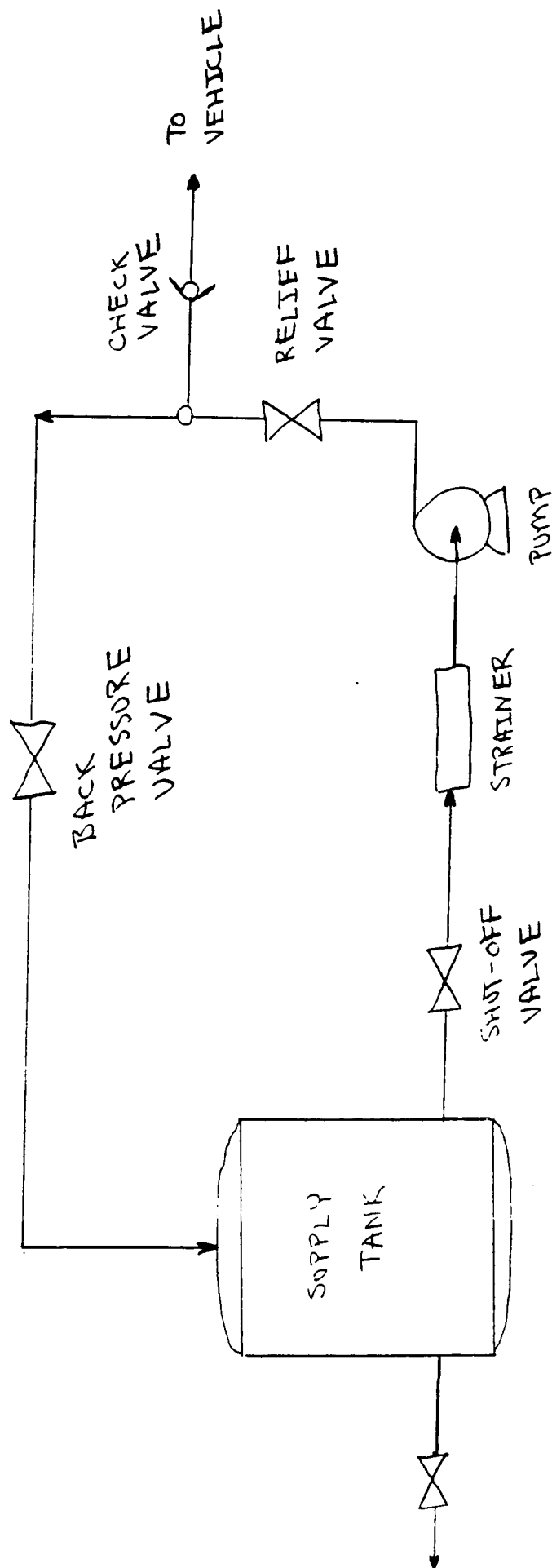


FIGURE IV
PROPULSION SYSTEM

APPENDIX 9

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LUNAR LAUNCH AND LANDING FACILITY, COMPLEX 391
NASA/USRA/UAOF/FTT

DECEMBER 3, 1987.

WATER RECOVERY SYSTEM

Chris Blatt

DEPARTMENT OF CHEMICAL ENGINEERING
FLORIDA INSTITUTE OF TECHNOLOGY
150 WEST UNIVERSITY BLVD.
MELBOURNE, FLORIDA 32901-6988

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WATER RECOVERY SYSTEM

Chris Blatt

DEPARTMENT OF CHEMICAL ENGINEERING
FLORIDA INSTITUTE OF TECHNOLOGY
150 WEST UNIVERSITY BLVD.
MELBOURNE, FL 32901-6988

ABSTRACT

The two issues that will be addressed here are those of the design of the water recovery system and the time it will take to dissipate the heat from each launch. The manner in which the water recovery system was designed was based on the total amount of heat that needed to be transferred so as to make the incoming vapor into liquid. From enthalpy differences and the amount of water vapor that will be entering the system, it seems that 100 MJ of heat must be transferred in order to accomplish the desired goal. From this data and the appropriate heat transfer equation, the design of the water recovery system was established.

pipes - 10

radius - ~~0.1~~ m 0.2m.

length - 50 m

8 internal pipes to the circular ring and 2 external pipes to the collection vessel, each 50 m in length.

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The second part of the design was to establish the time required to cool the pipes. The unsteady - state heat transfer equations were used to determine how long it would take. This value came out to be 67 hours.

Therefore, it seems possible to launch as many vehicles as desired as it will most likely take at least a few days to set up for the next launch. This is a good feature as the launches will not be held up by the thermodynamic inability of the water recovery system.

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WATER RECOVERY SYSTEM

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WATER RECOVERY SYSTEM

Chris Blatt

DEPARTMENT OF CHEMICAL ENGINEERING
FLORIDA INSTITUTE OF TECHNOLOGY
150 WEST UNIVERSITY BLVD.
MELBOURNE, FL 32901-6988

1. INTRODUCTION

The NASA/USRA University Design Program is a unique national program that brings together NASA engineers and students and faculty from U.S. engineering schools by integrating current and future NASA space and aeronautics engineering design projects into the university engineering design curriculum. The Advanced Space Design Program was conceived in the fall of 1984 as a pilot project to foster engineering design education in the universities and to supplement NASA's in house efforts in advanced planning for space and aeronautics design. Note that the term "Advanced" was defined as being post Space Station Initial Operating Configuration. Nine universities and five NASA centers participated in the first year of the pilot project. Close cooperation between the NASA centers and the universities, the careful selection of design topics, and the unbridled enthusiasm of the students resulted in a successful first year and the decision to extend the experiment to another year. This

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second year brought nineteen universities and eight NASA centers into the program and saw the formation of the Advanced Design Program for Aeronautics. As of 1985, there were thirty-one universities participating in the project.

Each university is assigned a certain topic in which they must work out the details of design project. Florida Institute of Technology has been assigned the project of designing the LUNAR LAUNCH AND LANDING FACILITY, (COMPLEX 39L). This project includes many aspects of the design ranging from the lunar lander to the service and control facilities that will be needed to operate the lunar base. Each topic can be designed as a unit and then integrated into the overall design project so as to establish how effective and operative the system might be in actual operation.

The specific unit that will be addressed here deals with the recovery of the water vapor from the lunar launch site. That is, as an attempt to conserve as much as possible in the way of resources and to minimize possible environmental damages to the lunar surfaces and structures, it seemed worth while to investigate the possibility of designing such a recovery system. The design itself will mostly include the sizing and shape of the recovery system and the time required between launches.

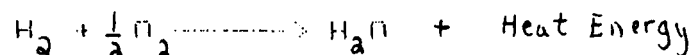
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2..

WATER RECOVERY

2.A. General background

Due to the low gravitational field that exists on the moon there is very little atmosphere, and thus no free water or other essential gases. It looks as if there is trapped oxygen in the lunar rock and this can be recovered to yield a valuable supply of oxygen for lunar activities. Hydrogen, however, must be obtained from another source; that source being the earth. With liquid hydrogen and liquid oxygen to be used as fuel for the space vehicles, it is possible to launch many space ships from the lunar surface. Notice that in the process of launching, the hydrogen combusts with the oxygen to produce water vapor.



Since water is so scarce, it makes sense to attempt to trap this water and use it in the space facilities (lunar base, etc.) that will be situated on the moon. The general procedure for this process of water recovery could be outlined as follows:

- trap the hot vapors from the launch site
- slow and cool down the vapors

- condense the vapor to liquid form
- have the liquid water accessible to the lunar facilities

2.B. The Design Equations and Results

Let's start the recovery procedure by taking a look at the actual trapping of the vapor. It seems that a confined space is required for the vehicle upon launch so that the vapors will not disperse along the lunar surface. The most reasonable solution to this would be a verticle silo that could contain the vehicle before and during the initial launch sequence. This silo would be below the lunar surface so as to keep the vapors in a confined space. The actual design of the silo itself will be mostly like that of missile silos that are present on the earth. The length of the silo will be 30m while the diameter will be 12m. A brief diagram of this can be seen in Figure 1.0.

Now that the vapors are in a confined space, it is necessary to cool and funnel the vapors to the collection spot. This will be accomplished by eight 50m pipes that will lead to a large circular ring that will surround the entire launch area. Notice that the bottom of the silo is widened out so as to allow for the connection of the several condensing pipes that will run from the silo to the collection area. The ring will not only provide more surface area for the vapor to condense on, but will also reduce the velocity of the vapor molecules by changing their momentum. This will occur by the molecules hitting the ring itself, and also by encountering more of

the vapor that is coming in the opposite direction. Remember that velocity is a form of energy and the more energy that can be taken away from the molecules, the quicker they will condense. Thus, it makes sense to slow the molecules down as much as possible as they are being cooled by the surrounding pipes.

Now, the ring will have two 50m external pipes that will be placed inbetween the internal pipes and will lead to a storage tank. These pipes will be spaced between the internal pipes to allow the vapor more surface area for heat transfer. Also, the vapors will have to change directions several times before entering these pipes and this will tend to reduce its velocity even more and thus, the energy of the vapor. After the vapor goes through the external pipes, it will be dumped into the storage tank. This storage tank is where the liquid water will be kept until it needs to be used by the lunar facilities (see figure 2.0).

The major concerns that are addressed deal with the amount of heat that must be transferred from the water vapor to the surrounding pipes in order for the vapor to condense upon reaching the holding area. For an initial estimate, the vapor will come in at 2900 K with an enthalpy of about 3.7 MJ / Kg. The end desired final state is water at around 373 C and 101.3kPa. This will have an enthalpy of 0.42 MJ / Kg. This suggests the heat that must be removed will be on the order of 3.3 MJ / Kg.

Since about 300 Kg of vapor per second is expected to enter the pipes during the launch sequence, it can be easily seen that around 100 MJ/s

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must be removed from the silo and the condensing pipes. The value for the amount of water vapor released depends on the particular type of engine used. Since there is no actual data available at this time, the amount expelled will be based on the Centaur rocket program.

The next logical step is to relate the total heat flow to the characteristics of the system. This can be done by several heat equations. One of these is listed below.

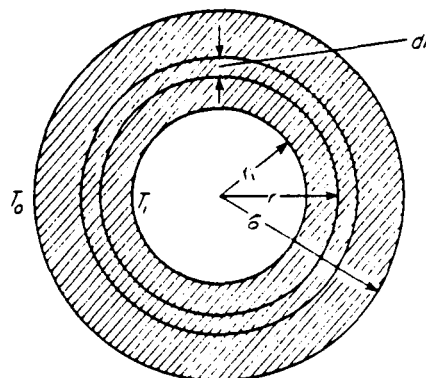
$$q = k * A * \frac{(T_i - T_o)}{R_o - R_i} \quad \text{eq\#1}$$

where:

- q - rate of heat transfer
- k - thermal conductivity of the material
- A - the log-mean area of the heat transfer area
- T_i - temperature of the inside of the cylinder
- R_i - radius of the inside of the cylinder
- T_o - temperature of the outside of the cylinder
- R_o - radius of the outside of the cylinder

SEE APPENDIX 1.0 FOR UNITS

Notice that this equation can only be used if the cylinder has a certain thickness as shown as follows:



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Since the pipes will be constructed of the same material as the surrounding rock, it is possible to conceive that the thickness of the cylinders could be 30m. This does not really seem reasonable as the rate of heat transfer is far too quick to consider such a thick wall for the cylinder.

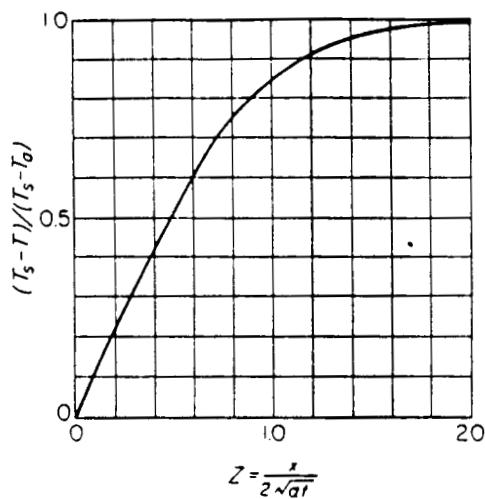
A reasonable approach to the thickness that should be used in equation #1 is the relationship between the time of heat transfer to the penetration distance into the surrounding wall. This distance will be a good approximation as to how thick the cylinder wall should be modelled. This equation is as follows:

$$\frac{T_s - T}{T_s - T_a} = \frac{2}{\sqrt{\pi}} \int_0^z e^{-z^2} dz \quad \text{eq#2}$$

where :

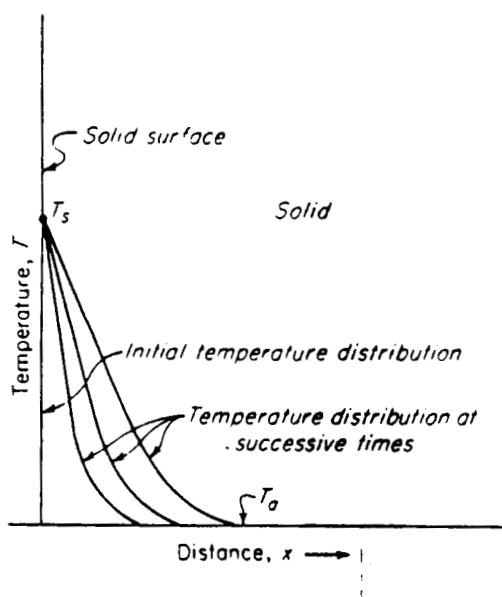
- $z = x/2 \sqrt{\alpha t}$ dimensionless value
- α - thermal diffusivity
- x - distance from surface
- t - time change in surface temperature

A graphical representation of the unsteady-state heat flow can be seen as follows:

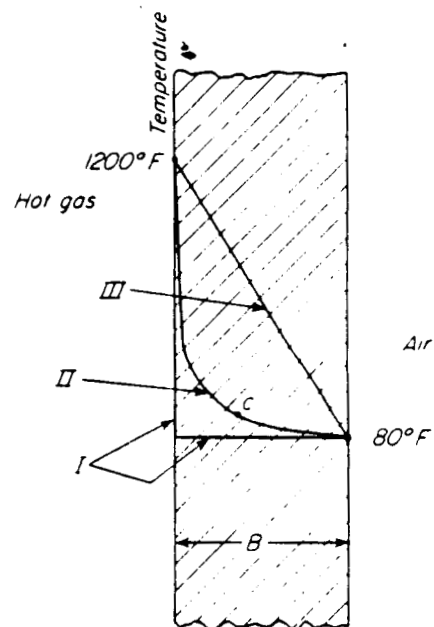


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(a)



(b)



(c)

Figure 4.0

To make an approximation of this integral, it is necessary to provide a desired temperature change. Since the heat flow will be very rapid,

the penetration distance will be reasonably small as there will not be much chance for the heat to diffuse away from the pipes. The desired temperature change will be chosen as one percent of the initial temperature change in the surface temperature. With this design basis, the following equation can be invoked.

$$X_p = 3.64 \sqrt{\alpha t} \quad \text{eq\#3}$$

where:

X_p - the penetration distance

α - the thermal diffusivity

t - time over which the heat transfer will occur

With the design basis of one second for the heat transfer rate, this yields a penetration distance of 5 cm.

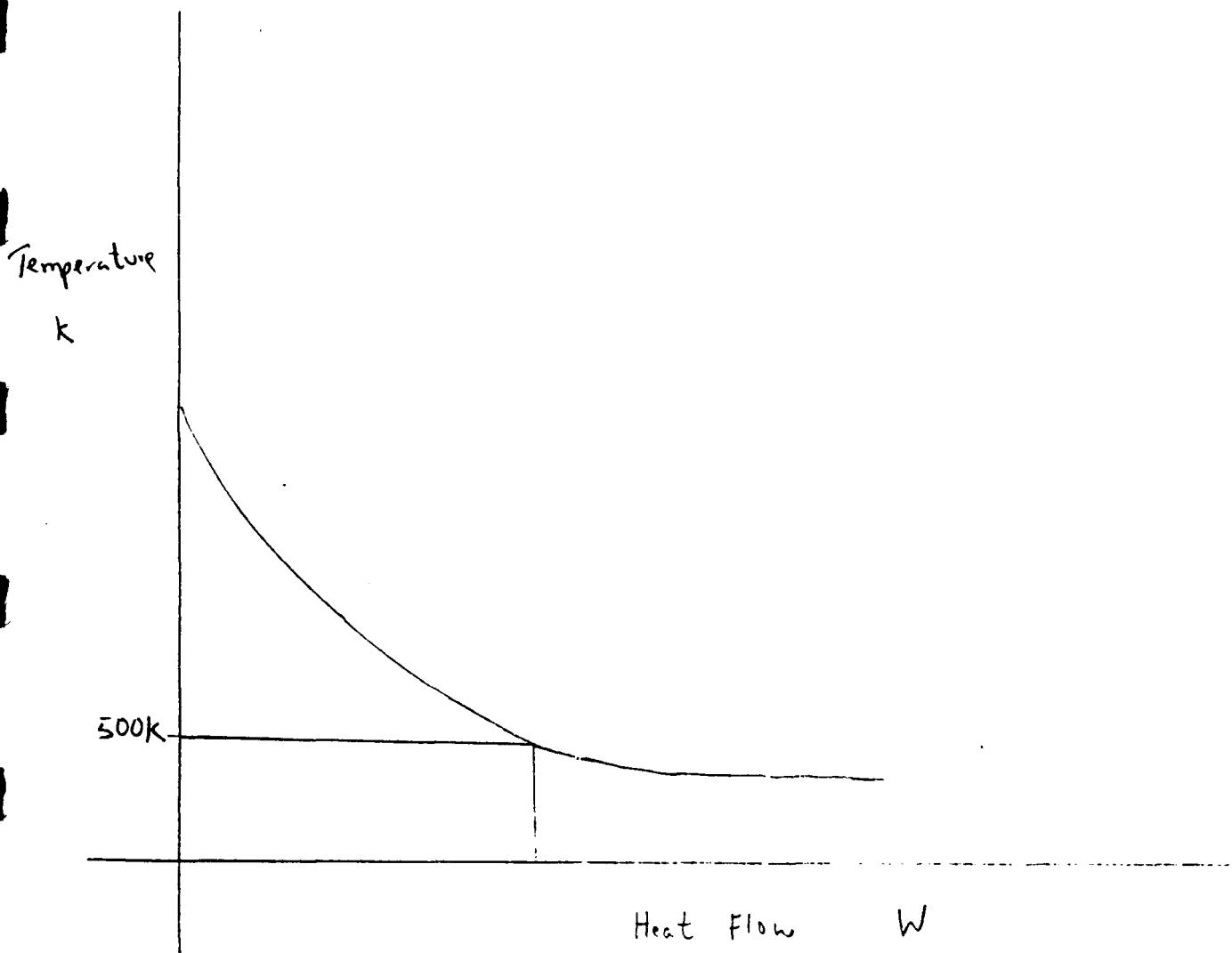
With the thickness of the cylinder wall known, equation #1 can be used to determine the design of the pipe. This requires a trial and error solution in order to find both the radius and length of the condensing pipes that will be used in the water recovery system. After much work, the values of .2m for the radius and 50m for the length of the pipe were established.

2.C. The Time of Cooling

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The next important aspect of the design is to establish how long it will take for heat to dissipate from the pipes to the surrounding lunar rock so that the temperature will return to approximately the same value as it was before the first launch. In order to arrive at a value for an acceptable temperature, the relationship between temperature and time of cooling must be established. Referring to equation #2, the relationship appears to be exponential. This will produce the following graph.

Temperature Vs Heat Flow in Unsteady State



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Now, a cut off point must be chosen in order to get a value for the acceptable temperature. From inspection of the graph, it is seen that beyond a certain time, the rate of temperature change is not as rapid as it was in the beginning. Therefore, this value is chosen from the graph to be 500K.

With this value, the time must now be established. This can be found by the following second order partial differential equation.

$$\frac{\partial T}{\partial t} = \alpha \frac{\partial^2 T}{\partial x^2} \quad \text{eq\#4}$$

This is a very difficult equation to solve explicitly. An approximation of the solution for a cylinder can be used to model the results. This equation is as follows.

$$t = \frac{rm}{5.78 \alpha} \ln \frac{0.692 (T_s - T_a)}{(T_s - T_b)} \quad \text{eq\#5}$$

where:

t - time to cool to the desired temperature
 T_s- the average temperature of the surrounding surface
 T_a- the initial temperature of the surface
 T_b- the desired temperature of the surface

Using the appropriate values, the value for the time came out to be on the order of 67 hours. That is, it will take about 67 hours in order for the pipes to return to the same conditions as they were before the launch.

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CONCLUSION

From the study done on the water recovery system, it was found out that it was possible to transform the entering water vapor into usable liquid. The necessary equations in order to accomplish this were based on the unsteady-state heat transfer functions. The design of the recovery system will be of considerable use as far as cost considerations are concerned. The amount of water that will be recovered could save up to \$68 million on transportation costs by shipping the water from the earth to the moon.

The frequency of the launches should be of no real concern as it was determined that it would take around 76 hours for the pipes to return to the same state as they were before the launch. Therefore, it should be possible to launch as many payloads as possible from the launch site with only a small restriction as far as time considerations are concerned.

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FUTURE WORK

The next step after this would be to go to a more detailed design level. In that level the unsteady-state heat flow would be either modelled on a computer simulator or a prototype (i.e. an actual model for the system would be used) could be used to show how accurate the initial design level really was. It would be necessary to simulate actual lunar conditions, that is a low temperature and pressure environment. Then a surge of hot water vapor must be introduced into the system to see how much heat is actually transferred into the lunar rock. Using heat analysis devices and thermo-couples, it would be possible to get some idea as to how the rock was actually conducting the heat. If this proved to be feasible, it might be considered possible to actually capture a large percentage of the water vapor from the launch site.

Other possible variations would be a different type of design to see if other geometries beside the circular ring pattern would prove useful. It is also of interest to see if the vapor really need be condensed all the way to liquid as established here as a design basis. Perhaps less heat could be taken out initially so as to arrive at a lower temperature vapor that could be later condensed over a longer period of time. This would, of course, lower the heat transfer area

that would be required to produce the previously required results.

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APPENDIX

1.0 THE SYMBOLS USED IN THE DESIGN PROPOSAL.

- A - Area in question, m^2
- Cp - Heat capacity of the material, cal/g C
- e - Base of Napierian logarithms, 2.71828...
- k - Thermal conductivity, cal/cm s C
- L - Length of the cylinder, cm
- q - heat transfer rate, W
- r - Radius of the cylinder, cm
- ri - Inside radius
- ro - Outside radius
- Ta - Initial temperature, K
- Th - Average temperature at the end of a certain time
- Ts - Surface temperature
- t - Time, s
- Xp - Penetration distance, cm
- Z - dimensionless number
-
- α - Thermal diffusivity, cm^2/s
- ρ - Density, g/cm^3

2.0 THE DATA TABLES

In order to do any thermal analysis of the lunar material, it was necessary to get values for the thermodynamic properties of the lunar rocks. Several samples of lunar rocks have been retrieved from the moon by the early space missions. These samples were then used to determine the specific thermodynamic properties of the lunar surface. The results are as follows.

Major components of the lunar rock: - Silicon Dioxide
 - Aluminum Dioxide
 - Calcium Oxide

The thermodynamic data is as follows.

$$\alpha = 4.67 \times 10^{-3} \text{ cm}^2/\text{s}$$

$$k = 23 \times 10^{-4} \text{ cal/cm s } ^\circ\text{C}$$

$$C_p = 0.22 \text{ J/g } ^\circ\text{C}$$

$$\rho = 2.39 \text{ g/cm}^3$$

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3.0 THE CONDITIONS OF THE WATER AT THE TWO POINTS OF INTEREST

<u>THERMO PROPERTY</u>	<u>INLET</u>	<u>OUTLET</u>
TEMPERATURE (K)	2900	373
PRESSURE (KPa)	22053.01	101.3
ENTHALPY (MJ/Kg)	3.7	0.42
MASS FLOW (Kg/s)	300	300

TOTAL HEAT THAT NEEDS TO BE TRANSFERRED = 100 MJ.

With 100MJ of heat transferred, the entering vapor will become liquid by the time it reaches the collection vessel.

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4.0 COMPARISON WITH THE CENTAUR ROCKET

<u>PROPERTY</u>	<u>LUNAR VEHICLE</u>	<u>CENTAUR</u>
Take-off Mass	16,000 Kg	16,000 Kg
Propellant Mass	12,000 Kg	13,600 Kg
Propellant System	H2/O2	H2/O2
Thrust	212 KN	134 KN
Operating Pressure	Vacuum	Vacuum
Height	10 m	10 m

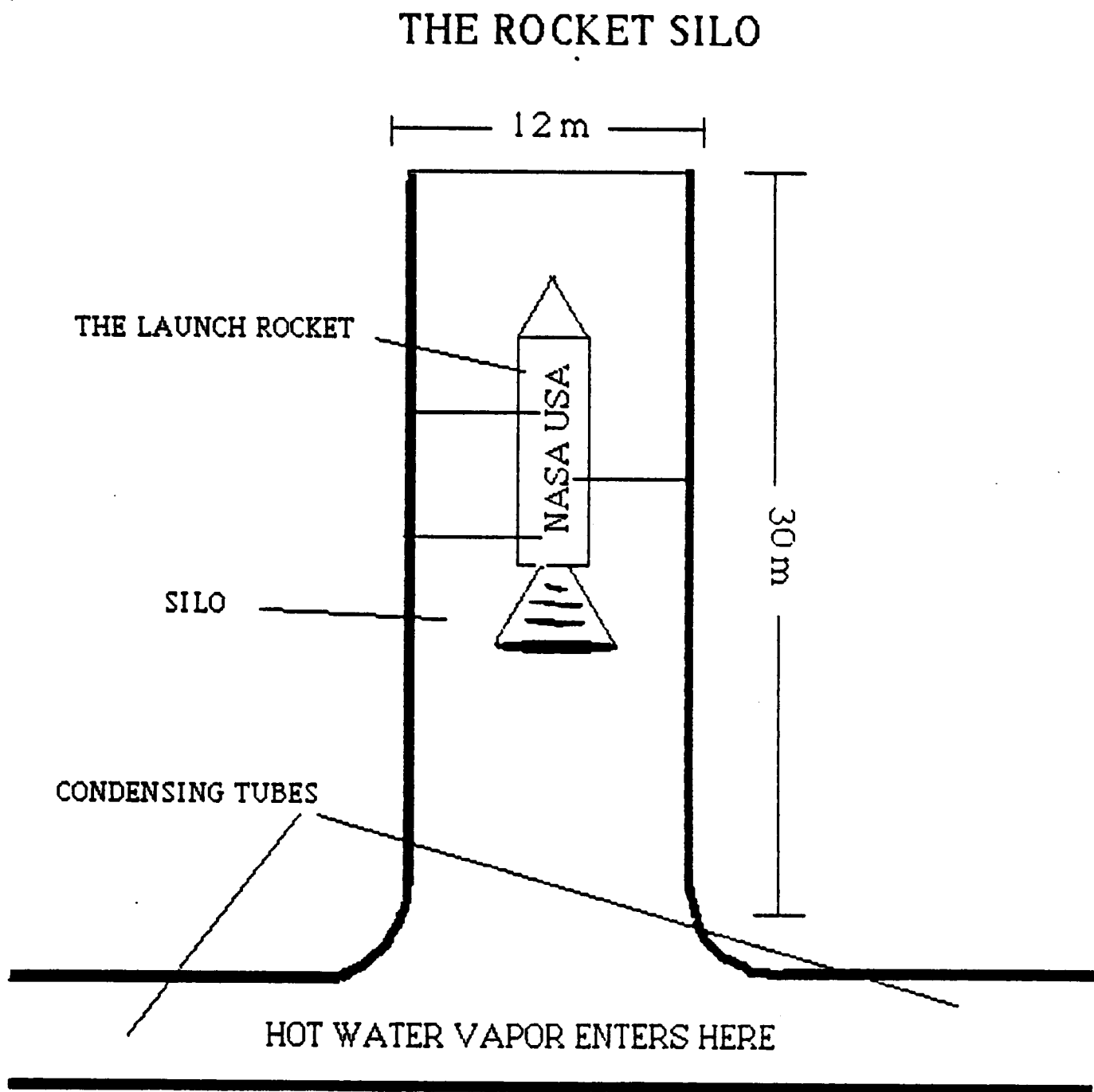


FIGURE 1.0

THE RECOVERY SYSTEM

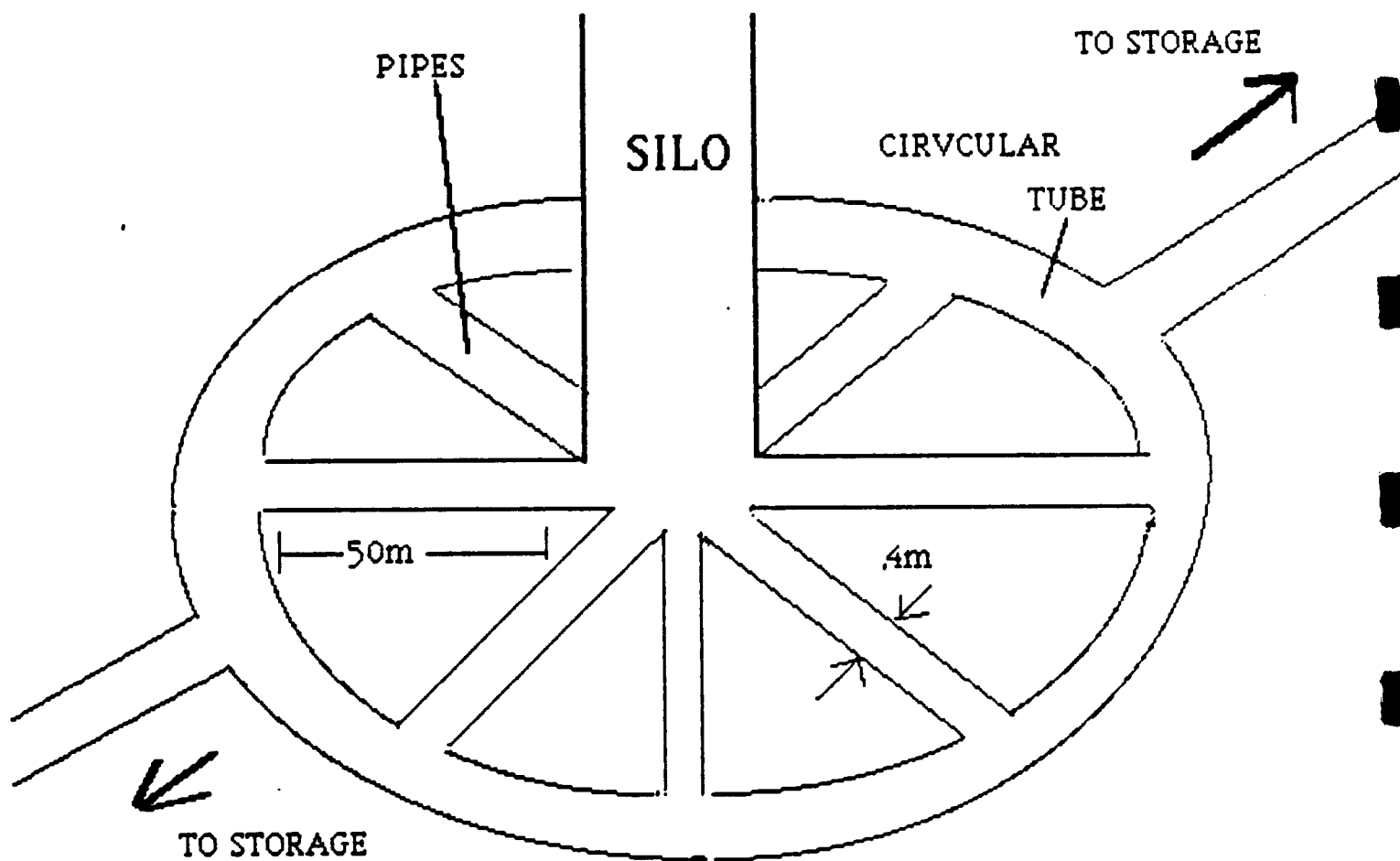


FIGURE 2.0

APPENDIX 10

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Lunar Landing Control Systems

by: Emily Hayes

EGN 4001
Advanced Space Design

December 3, 1987

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Lunar Landing Control Systems

ABSTRACT

The lunar lander control system will have probes, a radar detection system and television cameras. The probes will be extended from the footpads and the other two systems will be in the lander itself. Two different methods of landing are discussed. Recommendations are made for the landing mode and the landing site. Environmental impact is also taken into account and discussed.

LUNAR LANDING CONTROL SYSTEM

TABLE OF CONTENTS

1. Introduction
2. Equipment
 - A. Radar
 - B. TV Camera and VHF Antenna
 - C. Probes and Cables
3. Landing Modes
 - A. Soft Landing
 - B. Hard Landing
4. Landing Site and Environmental Recommendations
 - A. Landing Site
 - B. Environmental Factors
5. Cost
6. Closure

Lunar Landing Control System

1. INTRODUCTION

The lunar lander will have the set coordinates for a particular landing site. There will be two types of guidance systems: automatic and manual control. There will also be a set of four radar tracking stations on the moon located at various points to help in tracking the lander so that it can land in the appropriate place. The antennas for this must be carefully placed on the lander so they are not damaged in the docking stages.

There are two types of landings: hard and soft landings. Both of them were studied and the soft landing was shown to be the best alternative. Some problems exist for the soft landing also but they are not as great.

Another study was made on whether the landing site should be paved or unpaved. This study showed the paved site was the best alternative although it may be more expensive.

Finally the environment was taken into consideration. We tried to answer the question of how the environment would be affected and how the effects could be decreased.

2. EQUIPMENT

2.A. Radar

Four tracking stations will be located on the moon around the landing site which will be 250m in diameter maximum. The range of each of the stations will be approximately three to four miles. This range will allow the lander to determine any errors in guidance before it gets too close to the moon's surface. The radar antennas will be pointing inward from the outer parts of the landing site, but they will be able to rotate if necessary. It may be possible to decrease the number of tracking stations to three at 120° from each other in order to save money, but two would not be enough. It is recommended that the lunar lander be specially pointed so the extreme glare will not affect the radar.

This radar equipment will transfer the location of the lander into the lander computer. Then the computer may automatically control the landing on the surface. The size of the landing site may be reduced if the radar works well and the information can be transferred quickly enough to hit an exact point on the moon. The computer will have the exact coordinates in its memory, but in previous landings the lander has landed as far as 230m from the expected point. The problem lies in the timing delay from the radar to the lander and from the switch to the descent engine. The astronaut or computer will notice that a change needs to be made but there is a delay before the engine actually starts

The astronaut or coputer will notice that a change needs to be made but there is a delay before the engine actually starts turning the lander in the right direction. Because of the range covered by four stations being greater than the range covered by three, four stations will be worth the money so that the landing site may be decreased. If the landing site is decreased sufficiently, three stations would suffice.

2.B. VHF Antenna

This extra antenna is used for communications with the surface of the moon. This antenna will aid in radar if the transfer is not made to the lander computer due to the dust blown up during touchdown.

2.C. Probes

Another aid to the landing will be probes 1.6m in length which will be located below the footpads. When the probes touch the moon the engine will be shut down. These probes must be at least 1.6m in length so the engine delay time is accounted for. The probes will also aid in soft landings.

3. LANDING MODES

3.A. Soft Landings

Soft landing is when the lander uses its descent engine to hover over the moon in an upright position until the probes touch the surface. Then the engine is turned off to allow the lander to slowly descend to the moon. The different engines will be fired to change direction. The computer on board will have the altitude and direction measurements.

Most soft landings will be manually controlled, because when the probes touch the lander should not descend anymore. The engine is turned off and cables from the lander will be caught. These cables will be lowered from the footpads like the probes. The surface will have something to catch the cables and hold them to help lower the lander. Then the lander will be lowered to the moon. This landing will prevent alot of dust from being blown up. Also the lower gravity will be taken into account by the fact that the cables will aid in lowering the lander to the moon.

The soft landing is recommended because passengers and equipment will not be damaged in such a controlled landing to the moon.

Although the soft landing is the best method, some problems do exist. One problem is that the brakes will be used alot, and it could wear them out. Another problem with soft landings is that it is hard to control direction and attitude at such low speeds. It takes a long time due to delays to change direction. There is at least a 12ns delay to transfer information from the computer to the engine before it fires. At that point, the lander

speeds. It takes a long time due to delays to change direction. There is at least a 12ns delay to transfer information from the computer to the engine before it fires. At that point, the lander is closer to the moon and needs a further correction factor. This shows why it is very important for all directions and phase angles to be kept as close to the calculated values as possible. More corrections cause a point landing to be very difficult. The final problem is in connecting the cables to the ground. If a point landing can be accomplished this will not be a problem because they may be attached to specific places on the surface. Otherwise it may be necessary to use a controllable arm to grab the surface and pull the lander to the surface.

3.B. Hard Landings

In a hard landing an active control system controls the landing. The active system controls the impact of the vehicle. Accelerometers will be used to sense the velocity of the vehicle. The lander will directly descend to the moon from orbit. This type of landing would require more space for a landing site because it would not be completely stopped at the touchdown point. This type of landing would cause dust to be blown up by the engines and vibrations to occur due to the engines. This dust and these vibrations would cloud the radar tracking systems and cause difficulty in the final touchdown. This type of landing would also cause craters on the moon's surface if the landing site was unpaved.

Damage to equipment and to astronauts could occur. The vibrations or small quakes caused to the moon's surface by a hard landing could be very damaging if the frequency of landings was high. For these reasons a soft landing is recommended.

4. LANDING AND ENVIRONMENTAL RECOMMENDATIONS

4.A. Landing Site

There was a question as to whether the site should be paved or unpaved. An unpaved site would lead to problems because the dust would be kicked up when the engine gets close to the surface. Also repeated landings would cause craters to form on the surface and eventually the craters would grow together making the landing site inappropriate for landing. If a lunar station was built and the landing site had to be changed it would be very expensive. Therefore; it is recommended that the landing site be paved. Also a paved site would help in soft landings. The pavement would allow for a place for the cables to hook on to. The dust which is blown up would obscure the astronauts view and make the radar data unreliable. This would make it necessary to use inertially derived data to monitor automatic touchdown or as a basis for switching to manual control of the descent. Dual flight controls and windows, as well as gross attitude, attitude error, and vehicle rates information would be necessary due to the dust of an unpaved landing site.

4.B. Environmental Factors

The environment will be affected by the quakes of a hard landing. It could have the effects of a severe quake on the moon. Also the energy from the engines would alter the atmosphere with its gases. The shutting down of the engine before touchdown would lessen the amount of gases emitted into the air. Another factor is the causing of craters by landings on an unpaved site. To lessen the environmental effects, a soft landing on a paved site are recommended. It is also recommended to turn the engine off before touchdown.

5. COST

The cost will include the cost for four radar tracking stations, a radar antenna, a vhf antenna, a computer for calculations of velocity, attitude, and direction, and the ²⁵ probes and cables for a soft landing. Cost reductions are possible in having less radar tracking stations and less software for a soft landing. The minimum cost would be with a soft landing with a paved site due to expenditures necessary later for upkeep of any other combination.

6. CLOSURE

The proposal is for a soft landing with a paved site. Four radar tracking stations are recommended with a television, three probes, and a vhf antenna for lunar surface communication. The costs and environmental factors decided which would be the best possible landing modes and sites. The description of the landings are given. The problems are also mentioned, and further research must be done in those areas.

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